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AS-202
SPACECRAFT 011
PROPULSION PERFORMANCE EVALUATION
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Prepared Under Contract

NAS 9-4810

Prepared by

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FOREWARD

This program technical report is submitted to NASA/MSC in accordance with Task MSC/TRW 19A, contract NAS 9-4810. It contains the final propulsion performance evaluation of the Service Propulsion System of AS-202 and supersedes both TRW Report 05952-H018-R8-00, "AS-202, Service Propulsion System, Quick Look Analysis Report," dated 1 September 1966 and TRW Report 05952-6031-T800, "AS-202, Spacecraft 011, Propulsion Performance Evaluation," dated 19 September 1966.

The cooperation of the Propulsion Analysis Section of NASA/MSC and, in particular, the efforts of Mr. J. Thames in coordinating activities between NASA/MSC and TRW Systems and providing needed information has been greatly appreciated. Appreciation is expressed to the TRW Computation and Data Reduction Center contingent of the flight evaluation team for their diligent efforts in expediting the timely delivery of flight data.

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AS-202 PROPULSION PERFORMANCE EVALUATION

I. INTRODUCTION

This report is submitted in fulfillment of ASPO Task 19A, Subtask III, Final Postflight Performance and Malfunction Analysis Report, of Contract NAS 9-4810. Subtask III calls for "preparation and submittal of documentation which presents the results of final postflight performance and malfunction analyses within forty days after receipt of all necessary flight data". The minimum data required to perform a final propulsion performance analysis of the SPS from the flight of Apollo-Saturn 202 mission were received from NASA/MSC on 14 September 1966, requiring final propulsion evaluation input by 20 October 1966. Reference 1 was completed 19 September 1966 to meet this requirement. However, the key AS-202 SPS performance parameters have been re-evaluated due to changes of various data subsequent to the issuance of Reference 1, e.g., revised propellant gauging system biases for the dynamic effect of the propellant flowing through the zero-g cans and corrected thrust acceleration data scale factors and biases.

A significant effort by TRW Systems in the particular areas of telemetry review and detailed propulsion/propellant systems performance evaluation has been expended, the results of which are discussed herein. Also, included is a general discussion of the BEPP postflight evaluation program philosophy, flight test data used, SPS performance simulation, and the resulting propulsion/propellant systems performance parameters as derived from the postflight analysis.

II. SUMMARY

Mission AS-202 was flown from the Merritt Island Launch Area (MILA) on 25 August 1966. Flight data from four sources were available for the final analysis of the SPS discussed herein; i.e., data from the Kennedy Space Center (KSC), Antigua and Carnarvon tracking stations and from the Apollo Launch Data System (ALDS). The ALDS data were hardlined to the Manned Spacecraft Center at a reduced sample rate. The KSC data were used for a review of the ignition transients and first 35 seconds of the first SPS burn. The ALDS hardlined data and the Antigua data were used for the detailed analysis of the entire first SPS burn. The Carnarvon data were used for the detailed analysis of the second SPS burn and for evaluation of the third and fourth SPS burns. These data indicated that the service propulsion system functioned normally during the four burns.

The onboard recorder data from the second and subsequent burns was reviewed. This data was in a very preliminary status at the time of the review and was extremely noisy. No propulsion system anomalies were observed during the second, third and fourth burns from this data.

The Best Estimate of Propulsion Performance (BEPP) Program was used to determine the propulsion system performance parameters during the steady state portion of the SPS first and second burns. The BEPP Program utilizes input data from the guidance, the propulsion, and the propellant gauging systems and the reported propellant loaded and vehicle damp weights, engine acceptance tests, and vehicle configuration design data. The results from

Acceptance

Isp = 312.8

MR = 2

[redacted]

the final BEPP analysis of the SPS first burn indicate that the SPS engine thrust was 21,799 pounds, 279 pounds greater than the acceptance test value; the engine specific impulse was 311.5 seconds, 1.3 seconds less than the acceptance test value; and the engine mixture ratio was 1.95, 0.05 less than the acceptance test value. The results from the final BEPP analysis of the SPS second burn indicate that the SPS engine thrust was 21,863 pounds, 343 pounds greater than the acceptance test value; the engine specific impulse was 311.9 seconds, 0.9 seconds less than the acceptance test value; and the engine mixture ratio was 1.95, 0.05 less than the acceptance test value. The above performance parameters are given for the engine when operating with specification standard inlet conditions. Figures 7 through 14 present the derived propulsion performance parameter profiles for the engine operation during the SPS first burn and Figures 15 through 22 present corresponding profiles for the SPS second burn. The analysis technique, data used and these results are discussed in detail herein.

In addition to the propulsion system performance, the performance of the propellant loading, propellant gauging and the flight instrumentation systems were analyzed. The results of these analyses are also presented herein.

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III. TELEMETRY

A. Telemetry Data Acquisition

The flight data used in the Reference 1 analysis were transmitted over the Apollo Launch Data System (ALDS) hardline from the Antigua Ground Tracking Station to the Kennedy Spaceflight Center and relayed to the NASA Manned Spacecraft Center. In the transmission, data bits are normally dropped but not enough to significantly affect accuracy. At MSC the raw data were decommutated and calibrated into engineering units, producing the Phase I magnetic data tape. In the Phase 2 computer program, 26 of the functions for propulsion analysis were extracted at a maximum 20 sample per second rate, and the data tape was delivered to TRW. Here the data in the CDC 3600 computer format were reformatted for the IBM 7094 and processed to produce individual CalComp plots of each function. Custom processing of selected functions was necessary for the smoothing, differentiation, or integration required in various stages of the performance analysis. Only PCM data were handled in the ALDS. As the result, the measurements telemetered in PAM were not available for the quick-look analysis. Through the laudable efforts of the NASA/MSC personnel involved in reducing the ALDS data and also due to the minimum delay time features incorporated into the data reduction program, the flight test data was made available to TRW very quickly after the flight.

With the acquisition of Antigua data on tapes, the full complement of requested measurements was processed, including the PAM data. A notable improvement in the flight reconstruction matches to the test data has been achieved since the quick look analysis of the SPS first burn (Reference 2). This improvement was due to the inclusion of propellant temperature data and the use of propellant densities calculated therefrom.

The Antigua data which were significantly different from the ALDS data were used in the final analysis; otherwise the original data from the quick-look analysis were retained. The full data package at the original sample rates was processed from the Carnarvon data tapes for the second burn.

B. Telemetry Data Review

1. Thrust Acceleration Data

Output data from the Apollo Guidance Computer transmitted via the Apollo Launch Data System (ALDS) hardline were received as thrust velocities in orthogonal coordinates at one sample every two seconds during the first burn. The second burn data were extracted from the Carnarvon data tapes at the same sample rate. For both burns the thrust velocities in each coordinate were differentiated and the resultant acceleration profiles calculated. These acceleration profiles were used in the BEPP Program and in independent calculations of Isp. Since the publication of Reference 1, the guidance data were reprocessed to incorporate revised calibration information. These corrections caused significant differences in the BEPP derived parameters. An error analysis in the Guidance Acceleration Smoothing Program (GASP) showed the first burn data precision to vary from 0.040 ft/sec^2 to 0.088 ft/sec^2, one sigma. For the second burn, sigma varied from 0.040 to 0.064 ft/sec^2. Compared to the acceleration levels of 15 to 29 ft/sec^2 this error is relatively small. Because the redundancy provided in the guidance data downlink was carried through the ALDS transmission, it was possible to determine that no loss of information occurred in the telemetry and subsequent transmission of this data. This means no significant improvement in the basic guidance data was obtained from the Antigua data tapes for first burn.

2. Tank and Inlet Pressure Data

a. Propellant Tank and Engine Inlet Pressure Data Correlation

Calculations were made to correlate the telemetered tank pressures and engine inlet pressures. The results of these calculations indicated that the telemetered tank or engine inlet pressures were biased by as much as four psi during the first SPS burns and that the indicated tank pressure profiles were erroneous.

The engine inlet pressures were calculated from the telemetered tank pressures, nominal helium piping pressure drops, tank-to-inlet propellant feed line resistances (derived from acceptance tests), and flow rate profiles derived from the BEPP program. Corrections were made for the effects of acceleration on the hydraulic heads using the AGC acceleration data. The resulting calculated engine inlet pressures are compared to the corresponding telemetered values at several time points during the first burn in Table I. The pressure values in Table I show that the tank pressures and the engine inlet pressures are relatively consistent after 650 seconds (40 seconds after first burn ignition) but differ significantly earlier.

In order to determine which of the pressure profiles was correct, the BEPP program was run utilizing separately the propellant tank and the engine inlet pressures. The run which utilized the tank pressures only matched the acceleration data to within tank P 3.5 percent. The run utilizing the inlet pressures matched the acceleration profile to within inlet P 0.2 percent. Thus, it was concluded that the measured inlet pressures are more representative of the actual values.

A comparison of the data after shutdown indicates a 2 psia bias between oxidizer tank and inlet pressures and a negligible difference between fuel tank and inlet pressures.

Telemetry data indicates oxidizer and fuel tank pressures, and oxidizer and fuel inlet pressures during the 88 second burn have similar profiles and are very close in magnitude to those of the first burn. This is also true of the third and fourth burns.

b. Observations From Inlet Pressure Data

The inlet pressure increases (due to feed line pressure drops going to zero) at second burn shutdown are 20.0 psia for oxidizer and 24 psia for fuel compared with expected increases of 19.7 psia and 24.1 psia. The calculated flow pressure losses were based on acceptance test derived resistances. The increases in oxidizer and fuel inlet pressures at third and fourth burn shutdowns are the same as above. This indicates that the flow rates during these burns were approximately the same as those calculated for the first and second burns.

Analysis of the engine inlet pressures during the S-IVB burn indicates that the tanked oxidizer level was below the top of the standpipe while the level in the fuel tank was close to or above the top of the standpipe and the fuel standpipe was full of propellant. These conclusions were based on the following observations and calculations.

MCC-K data indicates a rise in oxidizer inlet pressure of 6.5 psia during the period from 410 seconds to S-IVB shutdown, and a drop of 14 psia in oxidizer inlet pressure at shutdown. This agrees fairly well with the calculated 6.2 psia rise due to the increasing S-IVB thrust acceleration and a 16.6 psia drop calculated for shutdown. This indicates that the initial oxidizer level was below the top of the standpipe and that the standpipe was empty.

At the most, a 1 psia rise occurs in fuel inlet pressure during the period from 410 seconds to S-IVB shutdown, compared with a calculated expected rise of 3.7 psia. No step drop occurs at S-IVB shutdown compared with a calculated expected drop of 10.6 psia. An explanation for this observed data is that the fuel standpipe was full or very nearly full, and the propellant level was slightly above the standpipe during the period in question. Noting Figure 1, with an initial fuel level slightly above the standpipe at the prelaunch 1-g condition, as the thrust acceleration increases during launch, a small additional amount of propellant will flow into the storage tank. The pressure equations are:

$$P_{u_1} = P_{u_2} - \rho_f \alpha h_1 \qquad \text{Equation 1}$$

$$P_{inlet} = P_{u_1} + \rho_f \alpha h_2 \qquad \text{Equation 2}$$

where:

P_{u_1} = sump tank ullage pressure

P_{u_2} = storage tank ullage pressure

ρ_f = fuel density

h_1 = difference in propellant level between the sump & storage tank

h_2 = difference in propellant level between the sump tank and fuel inlet

α = thrust acceleration

Because of the large storage tank ullage volume, small additional amounts of propellant would result in insignificant changes in P_{u_2}. An increase as great as 1 cubic foot in propellant in the storage tank would result in an increase of only approximately 1 psia, compared with a decrease of approximately 73 psia in the sump tank ullage pressure. Therefore, P_{u_2} in the above equations may be considered a constant. Substituting Equation 1 into Equation 2 results in

$$P_{inlet} = P_{u_2} - \rho_f \alpha h_1 + \rho_f \alpha h_2 \qquad \text{Equation 3}$$

Assuming only a small amount of propellant is in the storage tank, h_1 would be approximately equal to h_2; therefore,

$$P_{inlet} = P_{u_2} = \text{a constant}, \qquad \text{Equation 4}$$

under all thrust acceleration conditions during boost. The sump tank ullage pressure decreases by an amount equal to the increase in thrust acceleration fluid head.

If the initial level of fuel in the standpipe at launch was just below the standpipe's top and was not refilled from the sump tank, the fuel level depression in the standpipe would be approximately 2 ft at S-IVB shutdown. This is calculated from the approximate 0.08 ft^3 increase in sump tank ullage volume corresponding to the decrease in sump tank ullage pressure from launch to S-IVB shutdown. From Equation 3, this 2 ft depression in h_1 should result in a fuel inlet pressure increase of approximately 0.8 psia during the period of the observed data.

If the initial fuel height in the standpipe was significantly below the standpipe's top, again from Equation 3 the fuel inlet pressure would increase more than was observed.

3. Propellant Temperature Data

The propellant temperature data from the PAM telemetry were processed and used in this final performance analysis. They could not be used in the quick-look analysis of first burn because the ALDS data package did not include the PAM data.

The propellant temperatures were relatively constant on each burn with the exception of the oxidizer temperature at the main valve inlet (SP0041). The latter decreased about three degrees during the second burn. The relatively good agreement observed between the feed line and valve inlet propellant temperature measurements during Mission AS-201 was not repeated on Mission AS-202 as seen in the following status chart.

Time	Fuel Feed Line Temperature (SP0008) °F	Fuel Main Valve Inlet Temperature (SP0040) °F	Oxidizer Feed Line Temperature (SP0005) °F	Oxidizer Main Valve Inlet Temperature (SP00041) °F
AS-202 All First Burn	71.5	78.5	72.0	81.0
AS-202 All Second Burn	69.0	78.0	63.5	84 → 81
AS-201 at 1300 sec from Range Zero	65.0	68.0	67.0	64.0

Several inconsistencies are apparent in the feed line temperatures. First, the temperatures at first burn initiation were significantly below the launch ambient temperature.

Initial helium tank temperature was 84°F, for example, and the inlet temperatures were 78.5°F and 81.0°F for fuel and oxidizer respectively. Second, the feed line temperature on either the oxidizer or fuel side should be approximately equal to or slightly greater than the valve inlet temperature due to heat transfer to the helium pressurization gas. It can be seen in the status chart, however, that the feed line temperatures for <u>both</u> propellants on <u>both</u> first and second burns are significantly <u>lower</u> (by 7-18°F), than the respective valve inlet temperatures. On Mission AS-201 only 3°F differences were observed, which could be within the accuracy limitations of the instrumentation. As a result of the above observations, the main valve inlet temperatures, SP0040 and SP0041, were used to calculate the propellant densities for the BEPP Program input.

4. <u>Propellant Density Data</u>

The propellant mass profiles telemetered from the PUGS were based on nominal propellant specific gravities with corrections for propellant temperature. However, the telemetered mass data must be corrected by using the measured rather than the nominal propellant specific gravities. The initial reported oxidizer density value was 2.0 percent higher than the nominal density of "green" N_2O_4. This value was later corrected to being 1.3 percent lower than nominal. Both values are outside of the allowable specification limits on composition, and thus are considered highly unlikely. The nominal density for "green" N_2O_4 was used for this report. However, the changes in oxidizer density resulted in the necessity to make an appreciable number of computer runs. The following density relationships were used in this analysis:

$$\rho_f = 58.56 - 0.03184T + 0.00036(P - 14.7)$$

$$\rho_O = 95.20 - 0.07804T + 0.00072(P - 14.7)$$

5. Chamber Pressure Data

The chamber pressure depicted by measurement SP0661 appears to poorly represent the actual SPS 202 flight operating regime both in terms of level and time-functional characteristics. An evaluation of SPS 202 flight data has resulted in the following conclusions concerning the SP0661 data:

1. The indicated 7 psi transient (or "hump") during the first 20 seconds of SPS first burn is not real.
2. The steady state telemetered P_c is biased by approximately +3 psi during the 4 SPS burns.
3. A large portion (approximately 3% P_c) of the time-functional growth prevalent throughout SPS first burn is attributable to instrumentation drift.

Aside from normal transients, all SPS 202 flight measurements, save only SP0661, indicate chamber pressure to be approximately constant during the initial 20 seconds of the SPS first burn. Propellant tank and valve inlet pressure measurements fail to provide substantiation for this "hump". None of these four pressures exhibit the characteristics in this region which would be necessary to produce the SP0661 "hump".

No abnormalities in this area during the SPS first burn are observable in the IGS-indicated thrust acceleration data, indicating that no build-up occurred in thrust. While excessive thrust chamber throat area shrinkage could explain the P_c "hump", it would at the same time produce a thrust "hump" which is not observable in the IGS acceleration data. ?

Reasons for the erroneous measurements which occurred in SP0661 are suspected to be associated with sensitivity of the transducer to the flight environment.

The IGS indicated thrust acceleration data, as well as the data generated in the preflight trajectory simulation, indicate the SP0661 steady state chamber pressure level to be 2 to 3 psi too high for the four burns. The levels of the propellant tank and valve inlet pressure measurements also support this conclusion.

The SP0661 indicated chamber pressure increase vs time during the SPS first burn appears to be approximately twice that implied by the indicated tank and valve inlet pressures. This is further verified in a comparison between the SP0661 data and that generated in the preflight trajectory. In addition, the IGS thrust acceleration data fail to reveal evidence of the thrust growth indicated by the chamber pressure data from the first burn.

C. Gauging System Data Evaluation

Both the primary and auxiliary propellant gauging systems performed predictably on Mission AS-202. Both systems had appreciable biases in the indicated masses of both the fuel and oxidizer, which were primarily caused by improper input information and by the design characteristics of the system interfaces. The biases are discussed in detail in Section C. 1. The flow rates indicated by the auxiliary gauging system were slightly different from the flow rates indicated by the primary gauging system, as seen in Figures 3 through 6.

1. Propellant Mass Biases

The propellant mass biases for both systems were from two sources: (1) known differences between the total propellant on board and propellant tank level being used as input to the gauging system, and (2) a difference in level between the propellant in the tank proper and the propellant in the standpipe. In addition, a number of possible biases existed due to the difference in sensing techniques for the two systems. The latter biases are discussed individually in the sections concerning the individual sensors.

The errors in the input information contributed biases between indicated propellant weights and true total propellant weights of approximately 1575 pounds of oxidizer and 600 pounds of fuel. Of these biases, the known difference between gaugeable and total propellant onboard contributed 525 pounds and 260 pounds for the oxidizer and fuel, respectively. In addition, a small correction to account for the mass of propellant vapor in the ullage gas was required. The remaining biases were caused by the occurrence of significant fluid velocity head in the zero-g cans.

The zero-g can bias results in a difference in liquid levels in the tanks and the inside of the gauging system standpipes. The standpipe is in essence a manometer which balances the pressure at the boottom of the standpipe with a fluid head. Under non-flow conditions, this fluid head equals the level of propellant in the tank.

However, when the propellant is flowing, the fluid head in the standpipe is reduced by the dynamic head of the propellant flowing by the bottom of the standpipe. Unfortunately, the zero-g can design is such that the propellant has an appreciable velocity at this point.

This effect is clearly seen in the drop in level indicated by both the fuel and oxidizer primary gauging systems soon after the first engine start. The magnitudes of the zero-g can biases were equivalent to 1000 pounds of oxidizer and 350 pounds of fuel.

NASA/MSC initially reported the biases to be 350 and 200 pounds of oxidizer and fuel, respectively, during ground tests. These values were initially used in the BEPP Program for correcting the indicated fluid level during engine firing. However, analysis of this effect indicates that the reported value of the fuel bias at one g may be too high. The value of the biases can be calculated from the propellant flow rates, flow areas, and density as follows:

$$\Delta H = \frac{V^2}{2g} \quad \text{and } V = \frac{\dot{W}}{\rho A_c}$$

$$\Delta W = \rho A_t \, \Delta H = \frac{A_t \dot{W}^2}{2g\rho {A_c}^2}$$

where

ΔW = propellant bias

A_t = tank surface area

ΔH = change in height

A_c = flow area in zero-g can

$\dot{W}$ = propellant flow rate

V = propellant velocity in zero-g can

ρ = density

g = local value of gravity

As the zero "g" can drawings show that the fuel and oxidizer flow areas are identical, the ratio of the oxidizer to fuel bias can be calculated as follows:

$$\frac{W_{ox}}{W_F} = \frac{A_{tox}}{A_{tF}} (MR)^2 \quad \frac{\rho_f}{\rho_{ox}} = 3.1$$

As a result of the discrepancy between the reported values of the bias and the calculated ratio, the test data from White Sands was re-examined and new values of 495 pounds and 157 pounds for the oxidizer and fuel biases respectively were obtained by telecon with Mr. John Norris on 17 November 1966. As shown in the table below, these values agree with both the calculated ratio and the observed bias during the initial firing. These values of the biases were used to obtain the propellant quantities for this report.

	At Normal Earth Gravity		Start (First Firing)		
	Initial White Sands	Latest White Sands	Using Initial Reported Values	Using White Sands Reported Values	Indicated From Flight
Fuel bias (lbm)	200	157	415	330	350
Oxid. bias (lbm)	350	495	725	1030	1000

2. Primary Gauging System

During the initial four seconds of the first firing, the primary system was locked out and thus indicated preset initial weights of 14,600 and 7,400 pounds for oxidizer and fuel, respectively. After the lockout period, the indicated propellant masses dropped rapidly to 13,600 and 7,050 pounds (after extrapolation back to zero time) for the oxidizer and fuel, respectively.

This rapid drop was caused by the zero-g can dynamic flow bias discussed in the preceding section. The indicated oxidizer mass linearly decreased for the remainder of the first firing. The fuel, on the other hand, took appreciably longer to stabilize and continued to show small shifts in level over the first forty seconds, decreasing linearly after this point.

The indicated propellant flow rates after stabilization were 43.9 lb/sec of oxidizer and 21.60 lb/sec of fuel. After correcting the propellant masses for the measured density, and the zero-g can bias effect, the indicated flow rates were 46.0 lb/sec of oxidizer and 22.85 lb/sec of fuel. No correction was made for the oxidizer vaporization as the capacitance system should detect and indicate the vapor mass as well as liquid mass. The indicated masses at the end of the first burn were 5238 pounds of oxidizer and 2726 pounds of fuel.

At the start of the second SPS firing, both the oxidizer and fuel gauging system indicated the same propellant masses as at the end of the first burn. These values were constant during the four-second lockout period. At the end of four seconds, the oxidizer capacitance gauge reading decreased rapidly to the proper level and continued to decrease linearly for the remainder of the second firing. A slight bias of +30 pounds was observed when extrapolating the data back to the time of the second start. This bias may have been caused by oxidizer vapor passing from the sump tank into the storage tank.

The fuel capacitance gauge profile, however, was quite erratic, as illustrated in Figure 2. Immediately after the four-second lockout, the sensor indicated an increase in mass of approximately 300 pounds. After rising to this value, the capacitance gauge reading took an unusually long time to return to the expected slope. The fuel capacitance gauge indicated a number of erroneous changes in flowrate during the second burn. In addition, a 120 pound positive fuel bias (as compared to the auxiliary system data or to the BEPP results) was observed during the second burn.

The observed characteristics of the primary fuel gauging system are similar to the effects that would be anticipated from a time lag in either the liquid level in the stillwell or in the response of the capacitance system servo activator. No satisfactory explanation has been found for this behavior.

3. Auxiliary Gauging System

In order to properly utilize the auxiliary system data, it is best to determine the estimated times of the sensor firings. If the data is simply smoothed without regard to the sensor location, a bias error can result.

This approach has been used in obtaining the auxiliary system indicated mass versus time. A weight versus time curve was recreated using the data at the predicted value of sensor actuation. Due to the coarseness of the PCM sensor data (120 pounds of oxidizer per bit change and 60 pounds of fuel per bit change) the actual time of firing was uncertain. However, using the data immediately after a sensor firing to predict a most likely sensor firing time resulted in the total propellant curves presented in Figures 3, 4, 5, and 6. These mass versus time relationships were used as input to the BEPP Program.

IV. DETAILED ANALYSIS

A final analysis of the propulsion and propellant gauging systems performance during first and second burns was completed. The SPS specific impulse was derived from the first burn thrust acceleration data, and an analysis of the propulsion and propellant gauging systems performance during both burns was made using the BEPP Program. The results of the first burn analyses were refined through analysis of inconsistencies from different data sources, addition of accurate propellant density data, and improvement of the BEPP matches to the data. The performance values presented herein are considered the best estimate of the actual performance of the system based on the best available flight data.

This section gives a brief discussion of the data used in the detailed analyses and the results derived from their use.

A. Data Input to the BEPP Program

1. Data Specifically Applicable to Spacecraft 011

From the telemetered data, tables of thrust acceleration, propellant tank pressures, interface pressures, chamber pressure and propellant volumes were input into the BEPP Program. In this final analysis, the propellant temperature measurements, which were telemetered by the PAM system, could be included, whereas they were absent in the quick-look analysis. Propellant line resistances were calculated from the pressure drops, flow rates and propellant densities from the acceptance tests of the engine flown in Spacecraft 011. The initial throat area was measured as 121.42 sq. in. while the change in throat area with burn duration was estimated for a typical engine as discussed in the next section.

The weights input to the first burn BEPP analysis were the weight of the loaded liquid phase oxidizer, the weight of the loaded liquid phase fuel, the vehicle damp weight, and the vehicle total weight at ignition. The second burn analysis was initiated with the BEPP derived weight status at first burn termination. These values were calculated according to the weight summary in Table II.

2. Vehicle Class Data

Inasmuch as the Spacecraft 011 propellant tanks were not calibrated directly with liquid propellants, design data from the fabricating contractor were used to derive the tank volume versus height tables. The decrease in throat area with time (as shown in Table III) was calculated from static tests carried out under conditions of thrust, flow rates, etc. which represent flight conditions.

A non-linear analytical engine model containing constants derived from static test data was used as the source of the linearized model employed. Using the non-linear model, variable linear influence coefficients were derived to relate changes in the inlet quantities of interface pressures, propellant densities, chamber pressure and chamber throat area to the propulsion parameters of thrust and propellant flow rates.

B. Analysis Results

1. Specific Impulse

The specific impulse of the Service Propulsion System during the first burn was calculated directly from the Inertial Guidance System acceleration data, with a small correction for the thrust rise rate. At 725 seconds, the value was 310.4 seconds.

The thrust rise rate correction which was applied to the calculated specific impulse value was taken from the BEPP derived thrust profile. This correction amounted to 1.1 seconds. The thrust rise rate was also calculated from the chamber pressure and thrust chamber throat area change rates.

[redacted]

In this case the correction amounted to 22 seconds to result in an absurd specific impulse value of 335 seconds. This result confirms that the chamber pressure transient indicated by the telemetered data was erroneous.

The contribution of the guidance data noise to Isp uncertainty is 0.12 seconds, one sigma. The thrust rise rate contribution to the error is negligible since the correction is small in magnitude. Therefore, based on $\sigma_{Isp} = 0.12$ seconds, the probability of the actual flight Isp being in a 0.5 second band about 310.4 seconds is about 95%, assuming no appreciable bias error is present in the acceleration data. The corresponding instantaneous value in the BEPP run used to derive the performance parameters given herein was 310.45 seconds.

Unfortunately, the relative shortness of the second burn precluded a precise specific impulse determination from acceleration data. Values from 310 seconds to 314 seconds could be calculated depending on which data time spans were used in the analysis.

2. <u>BEPP Derived Parameters</u>

Time histories of the AS-202 SPS performance parameters for the first burn are presented in Figures 7 through 14. No significant deviations appear to exist between these results and those predicted in the pre-flight trajectory. Engine thrust at standard inlet conditions as determined by BEPP is approximately 1.3% (279 pounds) higher than reported in AGC acceptance test log for engine s/n 029. Specific impulse at standard inlet conditions is 0.40% (1.26 secs) less than acceptance test.

[redacted]

Chamber pressure as depicted by SP0661 was not matched in this analysis inasmuch as this measurement, as previously discussed, appears to poorly characterize the SC 011 SPS chamber pressure flight profile. Attempts were made to match the SP0661 data but the results proved meaningless. The thrust acceleration "mismatch" was far in excess of tolerable limits. The chamber pressure profile generated by BEPP using measured propellant valve inlet pressures shows no "hump" during the 19 seconds following ignition as did the SP0661 data.

A summary of the BEPP results for the first burn along with acceptance test and pre-flight measured quantities is presented in Table IV. The accuracy with which the BEPP program matched propellant quantities as a function of time is presented in Table VI.

Time histories of the AS-202 SPS stage propulsion parameters for the second burn are presented in Figures 15 through 22. No significant deviations appear to exist between these results and those predicted in the pre-flight trajectory. Engine thrust at standard inlet conditions as determined by BEPP is approximately 1.6% (343 pounds) higher than reported in AGC acceptance test log for engine s/n 029. Specific impulse at standard inlet conditions is 0.28% (0.88 seconds) less than acceptance test.

Chamber pressure as depicted by SP0661 was not matched in the analysis of the second burn for the same reasons as described on the first burn.

A summary of the BEPP results for the second burn along with the acceptance test and pre-flight measured quantities is presented in Table V. The accuracy with which the BEPP program matched propellant quantities as a function of time is presented in Table VII.

3. Cutoff Impulse

The SPS cutoff impulse was calculated from the velocity gain during cutoff as recorded by the guidance system. The average value of cutoff impulse was 10,400 lb_m-sec, which is within the specification range of 8,000 to 13,000 lb_m-sec and 1100 lb_m-sec below the average value obtained from integration of flight chamber pressure data. In the operational trajectory a planned value of 11,300 lb_m-sec was used.

The first and fourth burn cutoff velocity gain data from the guidance system were calculated in Reference 9 as 11.6 ft/sec and 13.9 ft/sec respectively, referenced to cutoff times of 62957.60 and 66202.55 seconds. From the BEPP analyses, the average vehicle weights during these periods were 29,780 lb_m and 23,100 lb_m respectively. Impulse is defined as the thrust-time integral as follows:

$$I = \int_{t_{c/o}}^{t_{F=0}} F \, dt \qquad (1)$$

Inserting F = ma, and assuming the mass is approximately constant during cutoff, the following is obtained:

$$I = m \int_{t_{c/o}}^{t_{F=0}} a\,dt = \frac{W}{g_o} \left(V_{t_{c/o}} - V_{t_{F=0}} \right) \qquad (2)$$

[redacted]

where

I = cutoff impulse (lb_m-sec)

F = thrust (lb_f)

t = time (sec)

m = total vehicle mass (slugs)

W = total vehicle mass (lb_m)

a = thrust acceleration (ft/sec^2)

g_o = conversion factor (lb_m/slug)

V = thrust velocity (ft/sec)

Using Equation (2) the cutoff impulse is calculated as follows:

First Burn ($t_{c/o}$ = 62957.60 sec)

$$I = \frac{29,780}{32.174}(11.6) = 10,700 \ lb_m\text{-sec}$$

Fourth Burn ($t_{c/o}$ = 66202.55 sec)

$$I = \frac{23,100}{32.174}(13.9) = 10,000 \ lb_m\text{-sec}$$

$$I_{AVG} = 10,400 \ lb_m\text{-sec}$$

In the case chamber pressure data are used, the thrust coefficient and the throat area are assumed constant during cutoff, and the relationship becomes

$$I = \int_{t_{c/o}}^{t_{F=0}} F dt = \int_{t_{c/o}}^{t_{F=0}} C_f P_c A_t dt = C_f A_t \int_{t_{c/o}}^{t_{F=0}} P_c dt \qquad (3)$$

[redacted]

where

C_f = thrust coefficient (unitless)

A_t = throat area (in^2)

P_c = chamber pressure (lb_f/in^2)

Using A_t = 120.42 and $t_{c/o}$ = 62597.60 for first burn,

$$I = 11{,}600 \ lb_m\text{-sec}$$

For second burn, A_t = 119.87 and $t_{c/o}$ = 66176.50

$$I = 11{,}400 \ lb_m\text{-sec}$$

The average is

$$I_{AVG} = 11{,}500 \ lb_m\text{-sec}$$

V. PROPULSION SYSTEM PERFORMANCE EVALUATION

A. BEPP Program Philosophy

The TRW-developed Best Estimate of Propulsion Parameters (BEPP) Program was used to determine the AS-202 SPS performance parameters. This program utilizes a weighted, least-square technique in conjunction with all of the available data from static test in addition to the physical laws which describe the behavior of the propulsion/propellant systems and their interaction with the spacecraft. From the various flight and static-test-derived data, the simulation calculates the time histories of thrust acceleration, propellant weight consumed, inert weight expended, and the propulsion system performance parameters. The simulation embodies complete error models for the various flight and static test data used as inputs. The technique is to determine the coefficients of the propulsion and propellant systems performance parameters in the error model that minimize the quantity

$$X^2 = \sum_{j=1}^{n} \frac{(Z_j^* - Z_j)^2}{\sigma_j^{*2}}$$

where,

X^2 = a function to be minimized

Z_j^* = a flight test data point

Z_j = value corresponding to the flight test data point calculated by the simulation

σ_j^* = a-priori estimate of the standard deviation of the data point

B. Types of Flight Data Used

The flight test data are divided into the following three classes:

(1) statistically matched constraints

(2) imposed flight data from the particular flight

(3) standard spacecraft class parameters

Class (1) data are those matched statistically in a weighted least-squares sense. These data consist of the following:

Thrust acceleration time history

CM/SM damp weight

Loaded oxidizer weight

Loaded fuel weight

Propellant volumes on board at discrete times

Class (2) data are those derived for each specific vehicle. They are used as input to the propulsion and vehicle simulation of BEPP and consist of the following:

SPS engine start and cutoff times

Propellant density time histories

Propellant interface pressure time histories

Propellant tank pressure time histories

SPS nozzle throat area time history

Propellant feed line and engine internal resistances

Class (3) data are the standard spacecraft class parameters used as input to the propulsion and vehicle simulations of BEPP. In this analysis the Class (3) data consisted of the following:

SPS engine mathematical model

Tables of oxidizer and fuel propellant heights as functions of tank volumes

Miscellaneous flowrate schedule (ablative nozzle flowrate, propellant vaporization flowrates, RCS flowrates, etc.)

VI. TABLES

TABLE I

ENGINE INLET PRESSURE CORRELATION

610 ignitation

ENGINE INLET PRESSURE	Time - seconds after range zero					
	614	630	650	750	800	826.13
OXIDIZER						
Telemetered, psia	154	156	156	156	156	156
Calculated, psia (from [illegible])	156	160	158 4	157	156	156
FUEL						
Telemetered, psia	149	150	152	152	152	152
Calculated, psia	155	154	153	151	151	152

TABLE II

SPACECRAFT 011

VEHICLE WEIGHTS

SPS FIRST BURN

Item	Weight Summary (lbs)	BEPP Input Weight (lbs)
Command Module	12061 ± 50	
Service Module	9633 ± 50	
SLA Ring	91	
Total Dry Weight	21785	
Vaporized Propellant	+44	
Total Damp Weight	21829	
SM/RCS Ullage Maneuver	-15	
Damp Weight at SPS Ignition	21814	21814 lbs.
Total Oxidizer Tanked	15125 ± 75	
Vaporized Oxidizer in Ullage	-42	
Oxidizer in Liquid Phase	15083	15083 lbs.
Total Fuel Tanked	7570 ± 30	
Vaporized Fuel in Ullage	-2	
Fuel in Liquid Phase	7568	7568 lbs.
Total Vehicle Weight at SPS Ignition		44465 lbs

SPS SECOND BURN

Item	Weight Summary (lbs)	BEPP Input Weight (lbs)
Additional Vaporized Propellant	+26	
Total Damp Weight	21840	
SM/RCS Maneuvers	-41	
Damp Weight at SPS Ignition	21799	21799 lbs.
Total Oxidizer in Liquid Phase	5196	5196 lbs.
Total Fuel in Liquid Phase	2726	2726 lbs.
Total Vehicle Weight at SPS Ignition	29721	29713 lbs.

TABLE III

SPACECRAFT 011

THROAT AREA VERSUS BURN TIME

SPS First Burn	
Time from Range Zero (62132 Z) (sec)	Throat Area (in^2)
611	121.42
621	121.38
631	121.30
641	121.23
651	121.17
661	121.10
671	121.04
681	121.00
691	120.93
701	120.90
711	120.85
721	120.81
731	120.76
741	120.73
751	120.69
761	120.66
771	120.62
781	120.58
791	120.55
801	120.51
811	120.47
821	120.45
831	120.41
841	120.39
846	120.36
SPS Second Burn	
3957	119.87 (Constant)
4045	119.87

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TABLE IV

SPACECRAFT 011

SPS FIRST BURN PROPULSION PERFORMANCE

BEPP PROGRAM SUMMARY

Quantity	Specification	Reported	Reported σ	BEPP Value	BEPP σ	Deviation (p-r)	% Deviation (p - r)/r
$\bar{\bar{F}}$ engine, lb	21,500	21,520	72	21,799	8	279	1.297
$\bar{\bar{Isp}}$, engine, sec	309.20	312.80	0.70	311.54	0.19	-1.26	-0.403
Isp, average, sec				310.42			
$\bar{\bar{MR}}$, engine	2.000	2.000	0.007	1.951	0.004	-0.0488	-2.442
MR, average				1.995			
$\bar{\bar{\dot{w}}}_o$, lb/sec		45.90		46.26	0.10	0.36	0.789
$\bar{\bar{\dot{w}}}_f$, lb/sec		22.91		23.71	0.09	0.80	3.492
$\bar{\bar{\dot{w}}}_t$, lb/sec		68.81		69.97		1.16	1.689
Wd, lb		21,814	24	21,814		0	0.001
Wo_i, lb		15,083	25	15,076	8	-6	-0.043
W_{fi}, lb		7,568	10	7,658	7	90	1.186
W_{ti}, lb		44,465	107	44,549	27	83	0.188

[redacted]

[redacted]

TABLE V

SPACECRAFT 011

SPS SECOND BURN PROPULSION PERFORMANCE

BEPP PROGRAM SUMMARY

Quantity	Specification	Reported	Reported σ	BEPP Value	BEPP σ	Deviation (p - r)	% Deviation (p - r)/r
$\overline{\overline{F}}$ engine, lb.	21500	21520	72	21863	20	343	1.594
$\overline{\overline{Isp}}$, engine, sec.	309.20	312.80	0.70	311.92	0.29	- 0.88	- 0.280
Isp, average, sec.				310.38			
$\overline{\overline{MR}}$, engine	2.000	2.000	0.007	1.950	0.006	- 0.050	- 2.526
MR, average				2.017			
$\overline{\overline{\dot{w}}}_o$, lb/sec		45.90		46.33	0.12	0.43	0.930
$\overline{\overline{\dot{w}}}_f$, lb/sec		22.91		23.76	0.10	0.85	3.726
$\overline{\overline{\dot{w}}}_t$, lb/sec		68.81		70.09		1.28	1.861
Wd, lb.		21799	24	21815		16	0.073
W_{oi}, lb.		5196	100	5182	8	- 14	- 0.262
W_{fi}, lb.		2726	100	2698	7	- 28	- 1.010
W_{ti}, lb.		29721	150	29696	27	- 25	- 0.085

[redacted]

~~CONFIDENTIAL~~

Table VI

SPACECRAFT 011

SPS FIRST BURN BEPP SENSOR DATA MATCH

Time vol σ
813. 65.33 .26
825. 59.61 .23

OXIDIZER SENSOR QUANTITIES

Time (Sec)	Reported Volume (Ft^3)	A-Priori Estimate of σ (Ft^3)	BEPP Derived Volume (Ft^3)
620.0 .432	164.61	0.65	164.63
650.0 .508	149.41	0.59	149.18
685.0 .540	131.36	0.52	131.18
710.0 [illegible]	118.45	0.47	118.29
740.0 .596	102.99	0.41	102.88
765.0 .628	90.09	0.36	90.05
795.0 .608	74.61	0.30	74.65
813.3	65.18	0.26	65.26
826.0	59.13	0.23	59.25

FUEL SENSOR QUANTITIES

Time (Sec)	Reported Volume (Ft^3)	A-Priori Estimate of σ (Ft^3)	BEPP Derived Volume (Ft^3)
650.0	120.47	0.49	120.35
670.0	112.28	0.46	112.20
690.0	104.10	0.42	104.03
710.0	95.90	0.39	95.86
735.0	85.67	0.35	85.66
755.0	77.48	0.32	77.49
775.0	69.31	0.29	69.33
802.0	58.25	0.24	58.30
825.0	48.85	0.21	48.90

[redacted]

[redacted]

TABLE VII

SPACECRAFT 011

SPS SECOND BURN BEPP SENSOR DATA MATCH

OXIDIZER SENSOR QUANTITIES

Time (sec)	Reported Volume (ft^3)	A-priori Estimate of σ (ft^3)	BEPP Derived Volume (ft^3)
3967.5	52.84	0.21	52.72
3980.0	46.30	0.18	46.21
3992.5	39.86	0.15	39.71
4005.0	33.39	0.13	33.24
4017.5	26.85	0.10	26.77
4030.0	20.15	0.08	20.34

Time (sec)	Reported Volume (ft^3)	A-priori Estimate of σ (ft^3)	BEPP Derived Volume (ft^3)
3970.0	42.99	0.18	42.68
3982.5	37.77	0.16	37.63
3997.5	31.57	0.14	31.55
4012.5	25.38	0.11	25.47
4027.5	19.20	0.09	19.37
4040.0	14.33	0.07	14.28

[redacted]

NOMENCLATURE

$\bar{\bar{F}}$ = Engine thrust at standard inlet conditions (lbf).

$\bar{\bar{I}}sp$ = Engine specific impulse at standard inlet conditions (sec).

Isp = Engine specific impulse at operating conditions (sec).

$\bar{\bar{MR}}$ = Engine mixture ratio, o/f, at standard inlet conditions.

MR = Engine mixture ratio, o/f, at operating conditions.

$\bar{\bar{\dot{w}}}_o$ = Oxidizer flow rate at standard inlet conditions (lb/sec).

$\bar{\bar{\dot{w}}}_f$ = Fuel flow rate at standard inlet conditions (lb/sec).

$\bar{\bar{\dot{w}}}_t$ = Total propellant flow rate at standard inlet conditions (lb/sec).

W_d = Damp weight of Spacecraft, including RCS propellant (lb).

Woi = Weight of oxidizer in liquid phase at SPS ignition (lb).

W_{fi} = Weight of fuel in liquid phase at SPS ignition (lb).

W_{ti} = Total weight of spacecraft at ignition (lb).

VII. FIGURES

Figure 1

REFERENCE FIGURE FOR PRESSURE EQUATIONS

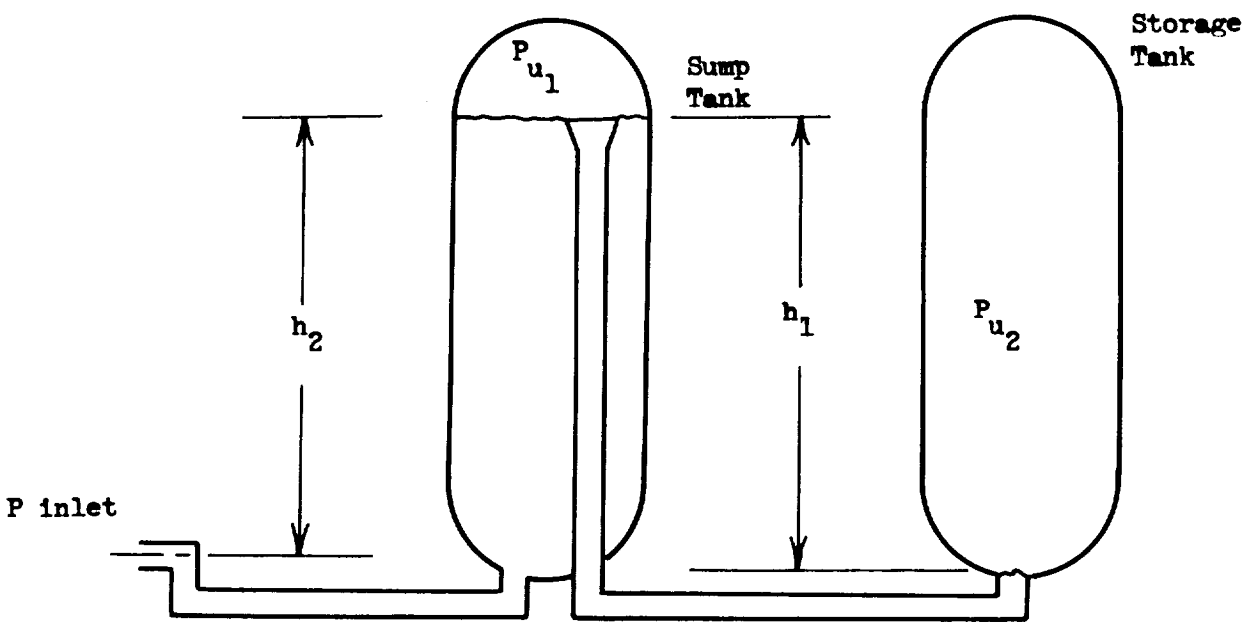

Figure 2

SECOND BURN ERRADIC OPERATION
OF
PRIMARY FUEL GAUGING SYSTEM

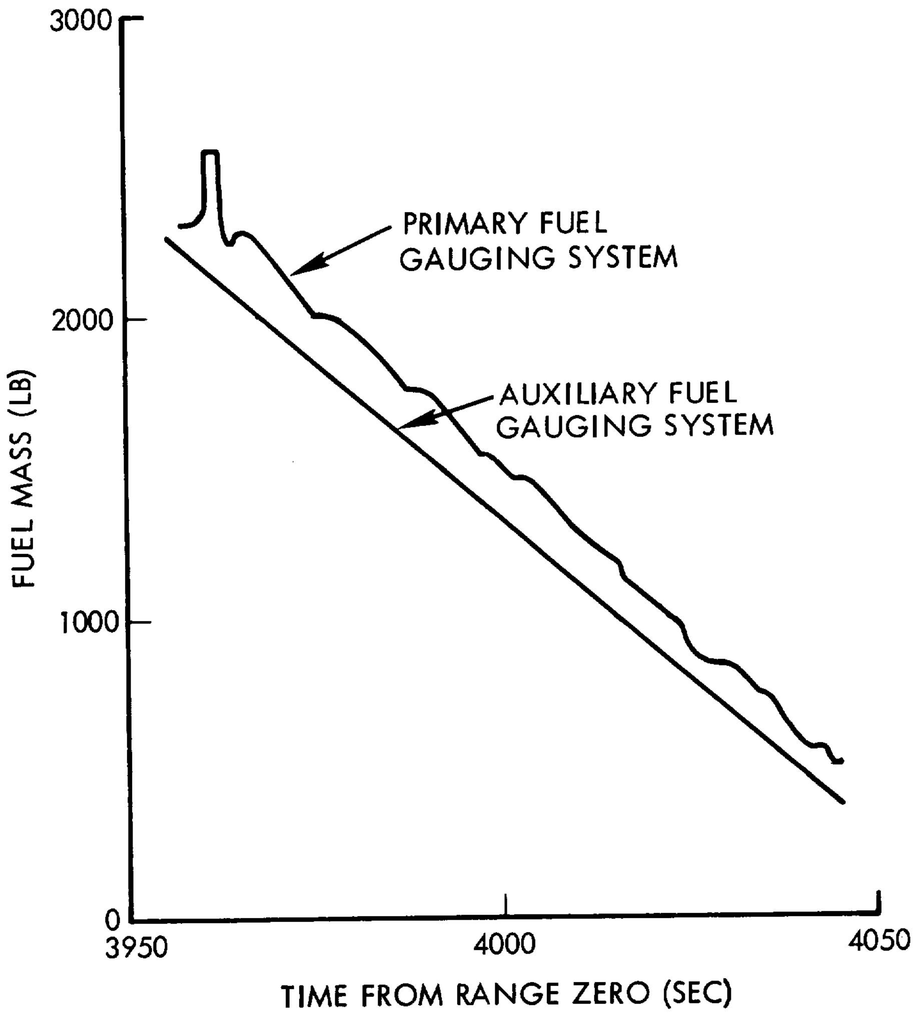

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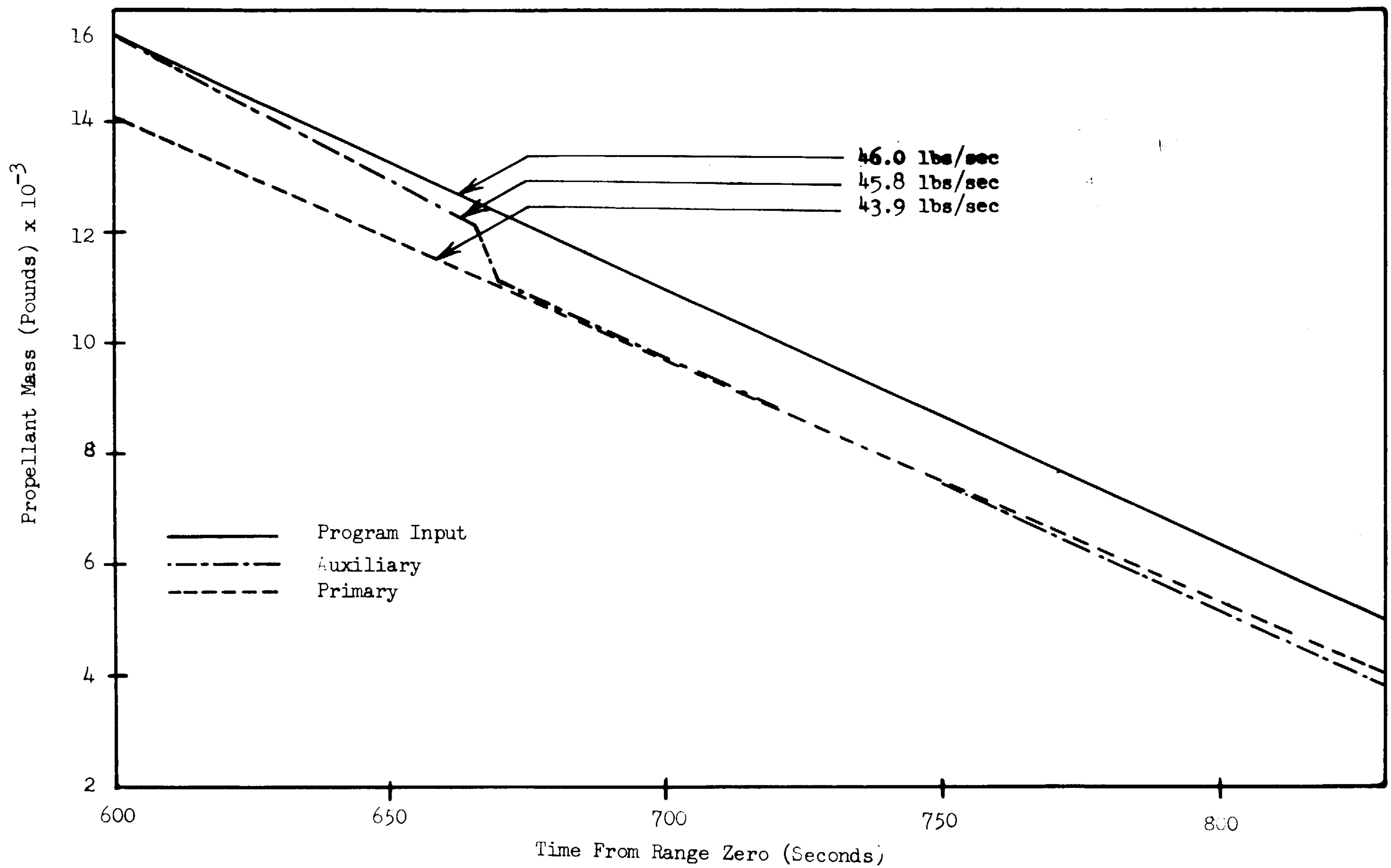
Figure 3

SPACECRAFT 011

SPS FIRST BURN OXIDIZER QUANTITIES DETERMINED FROM THE PROPELLANT GAUGING SYSTEM



[redacted]

[redacted]

Figure 4

SPACECRAFT 011

SPS FIRST BURN FUEL QUANTITIES DETERMINED FROM THE PROPELLANT GAUGING SYSTEM

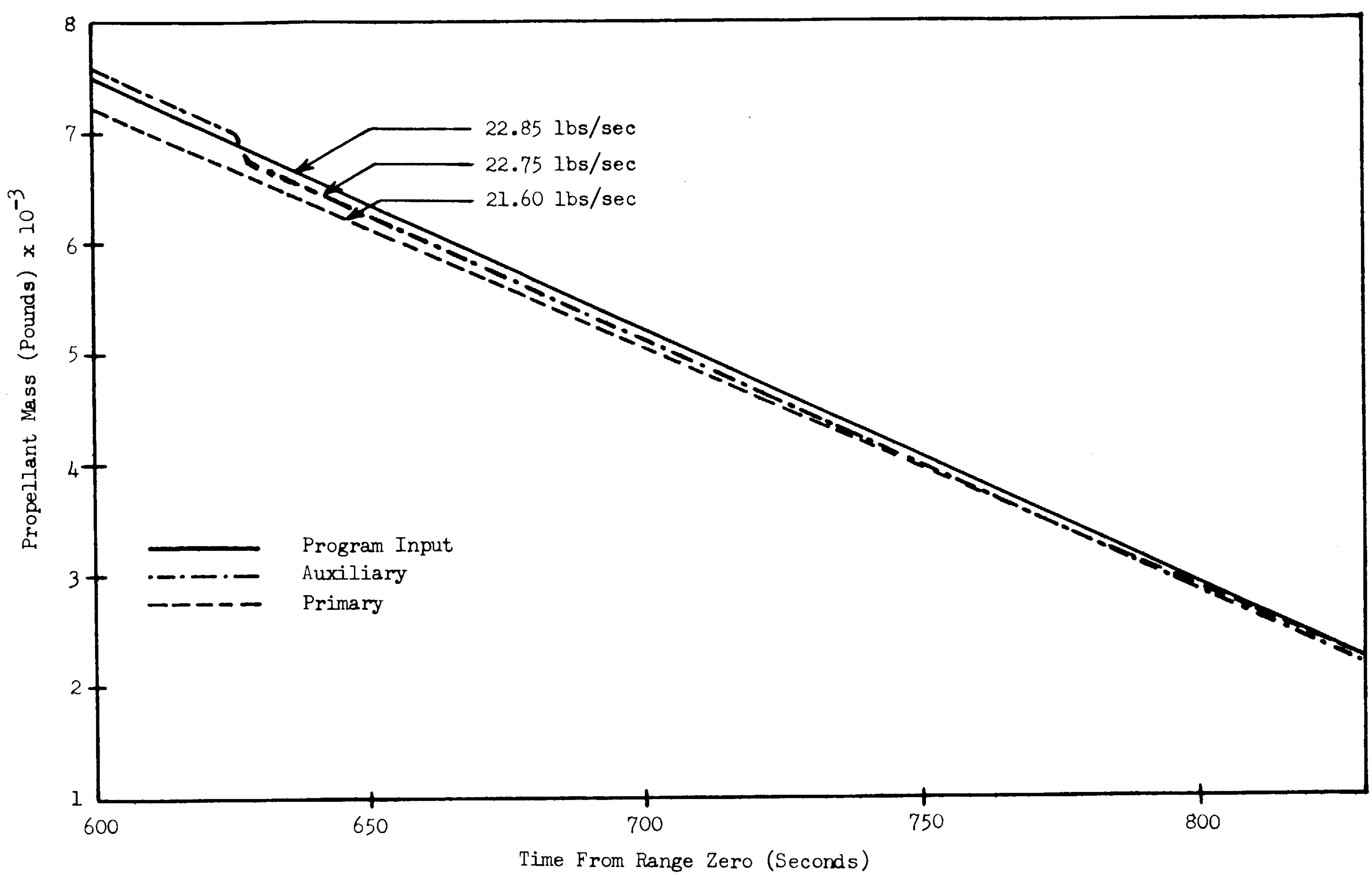


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Figure 5

SPACECRAFT 011

SPS SECOND BURN OXIDIZER QUANTITIES DETERMINED FROM THE PROPELLANT GAUGING SYSTEM

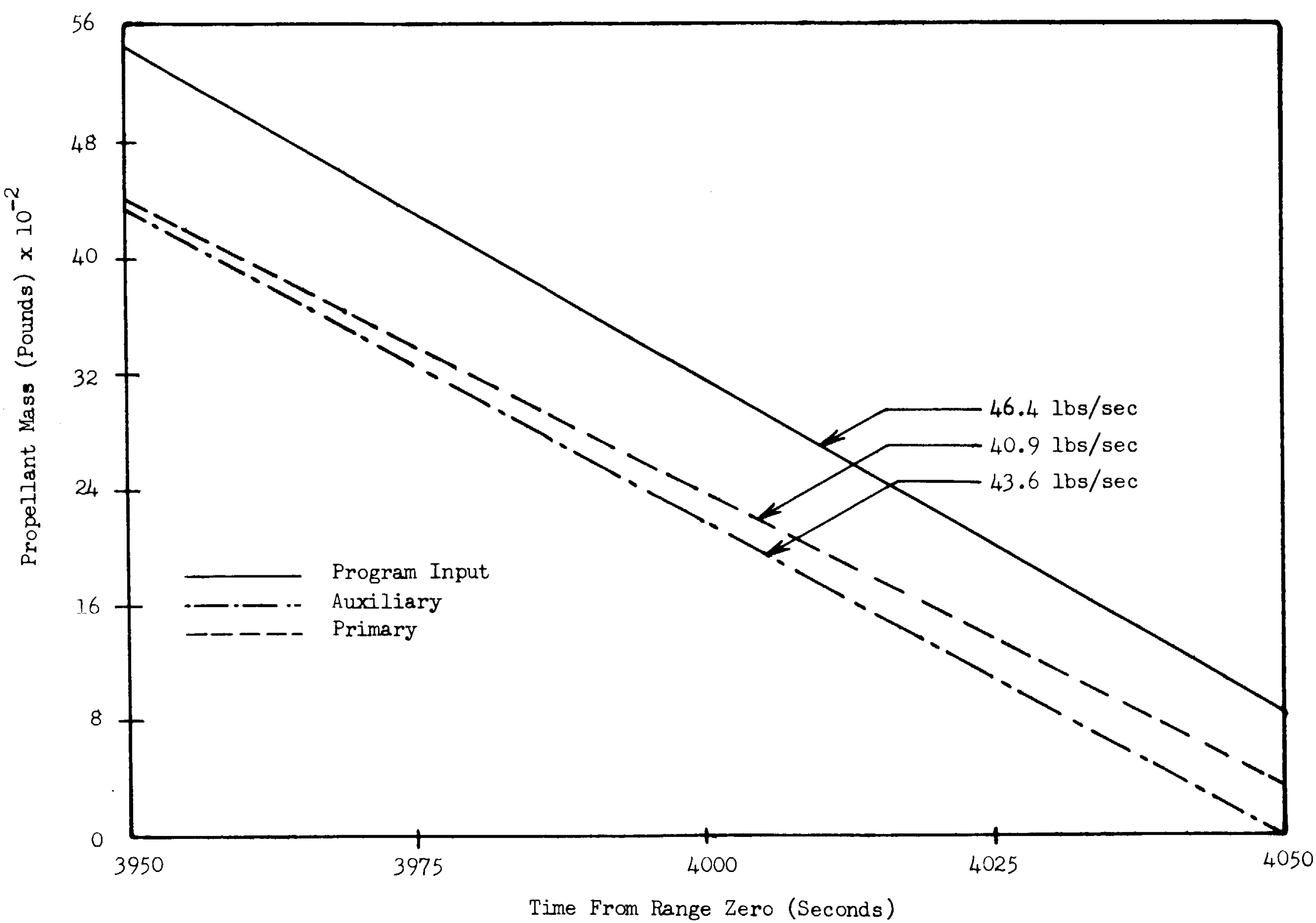

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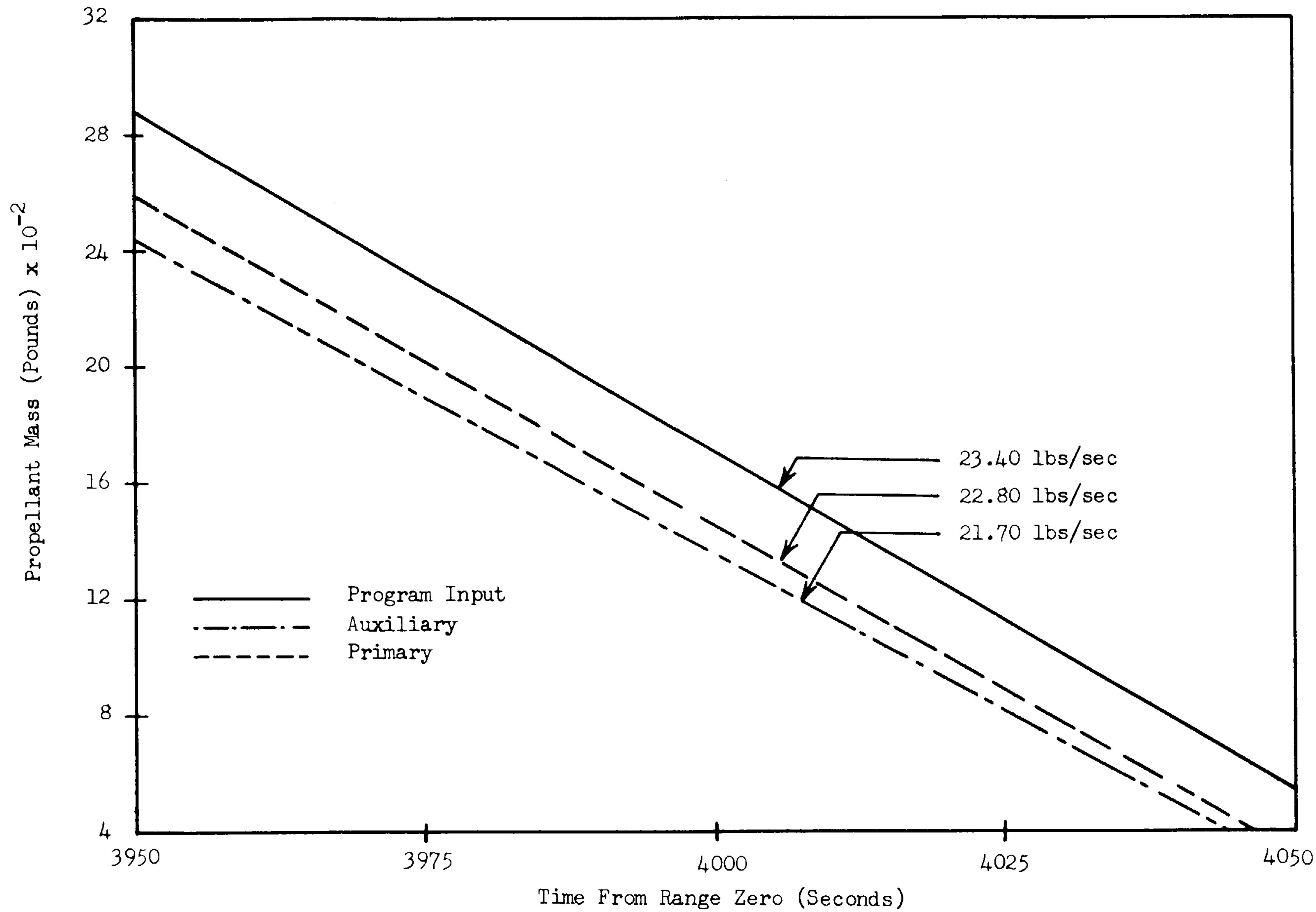
Figure 6

SPACECRAFT 011

SPS SECOND BURN FUEL QUANTITIES DETERMINED FROM THE PROPELLANT GAUGING SYSTEM



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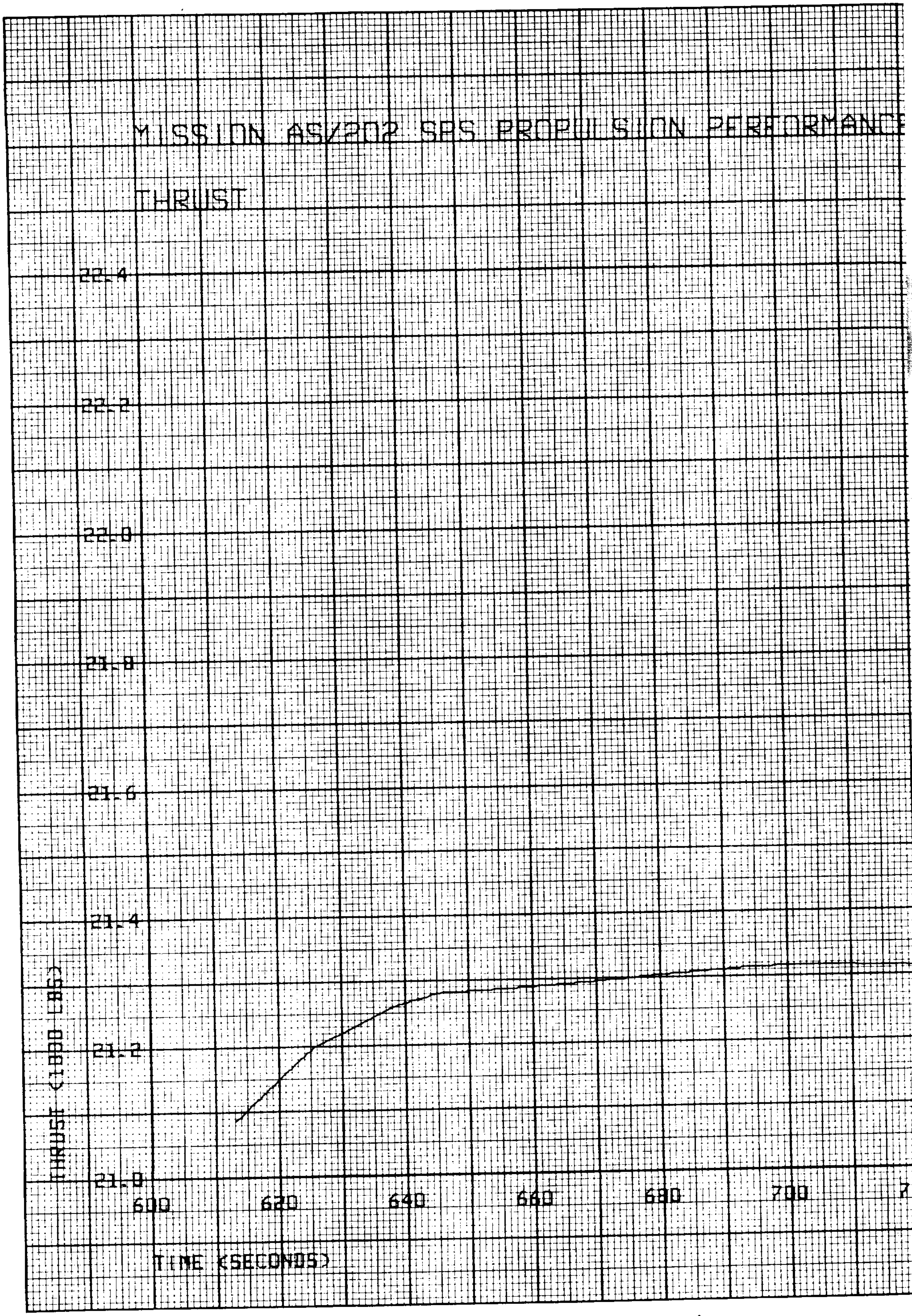

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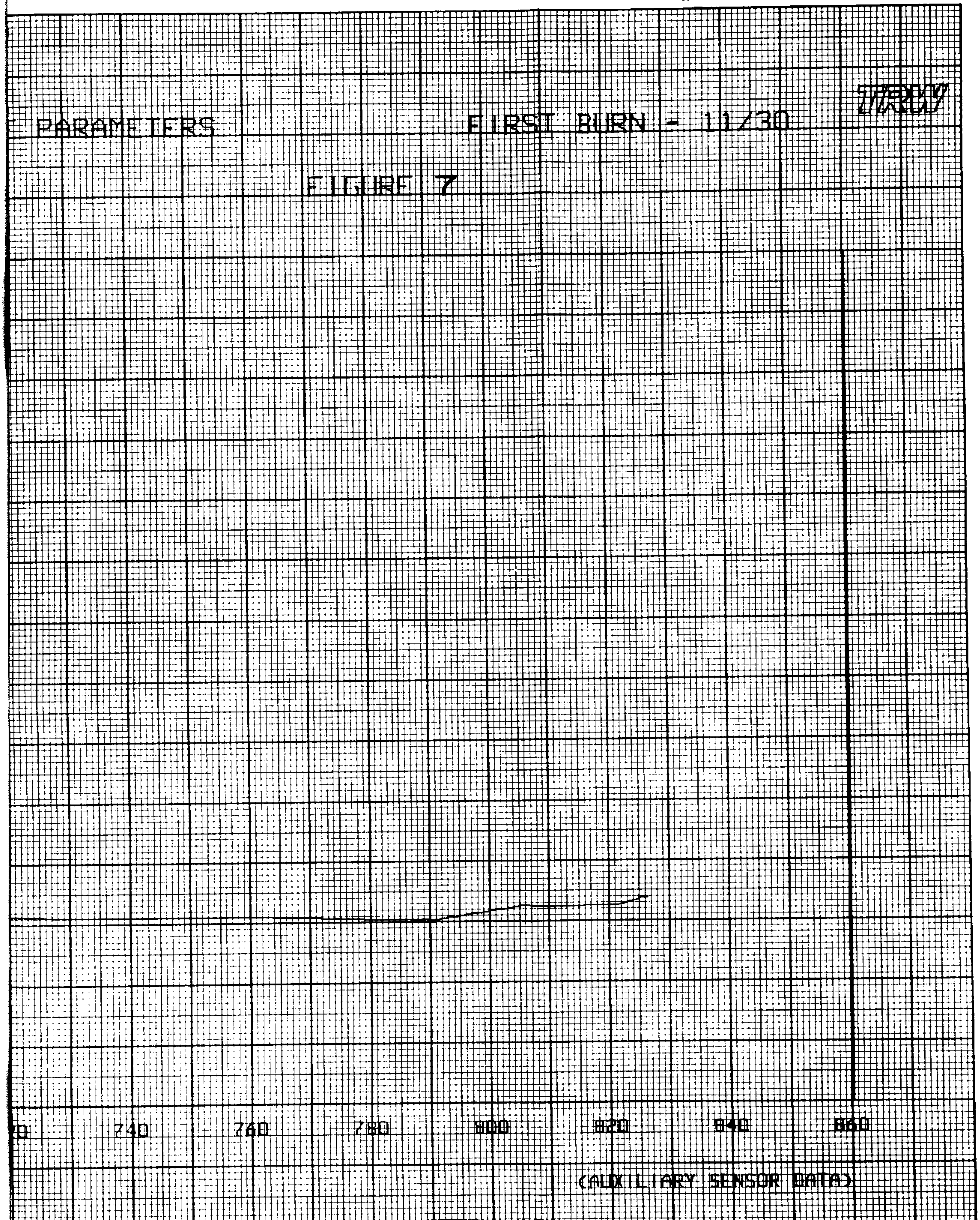

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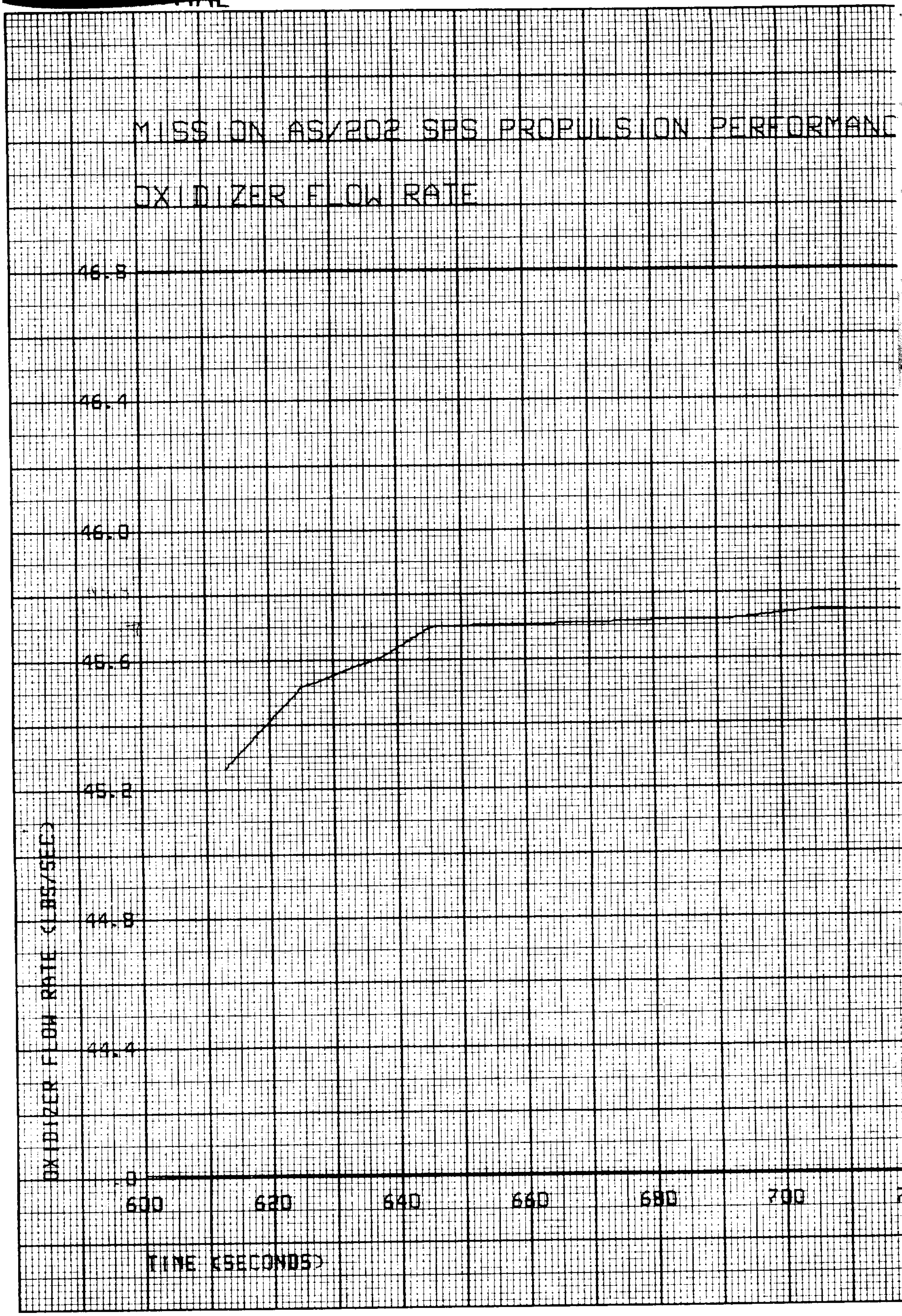

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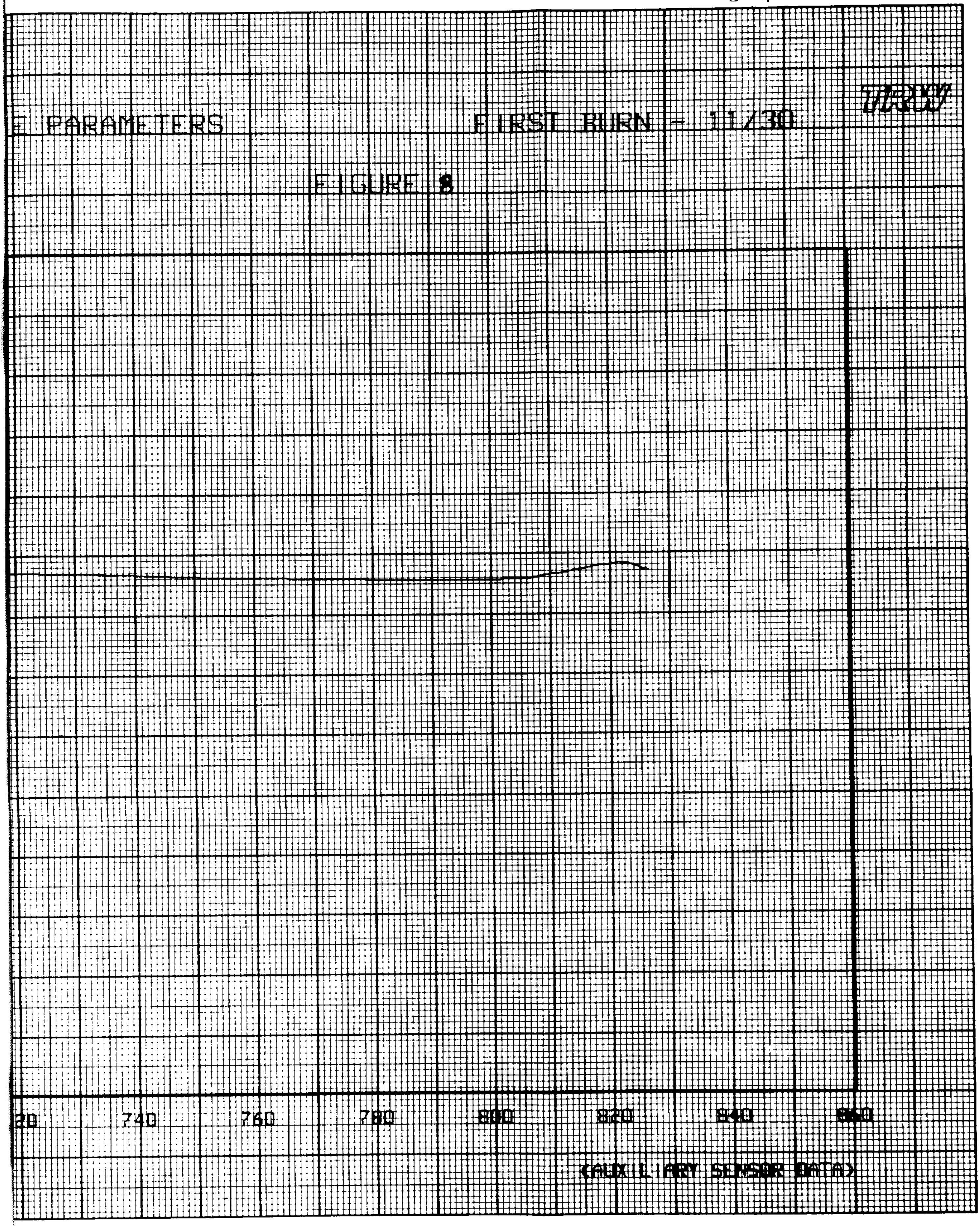

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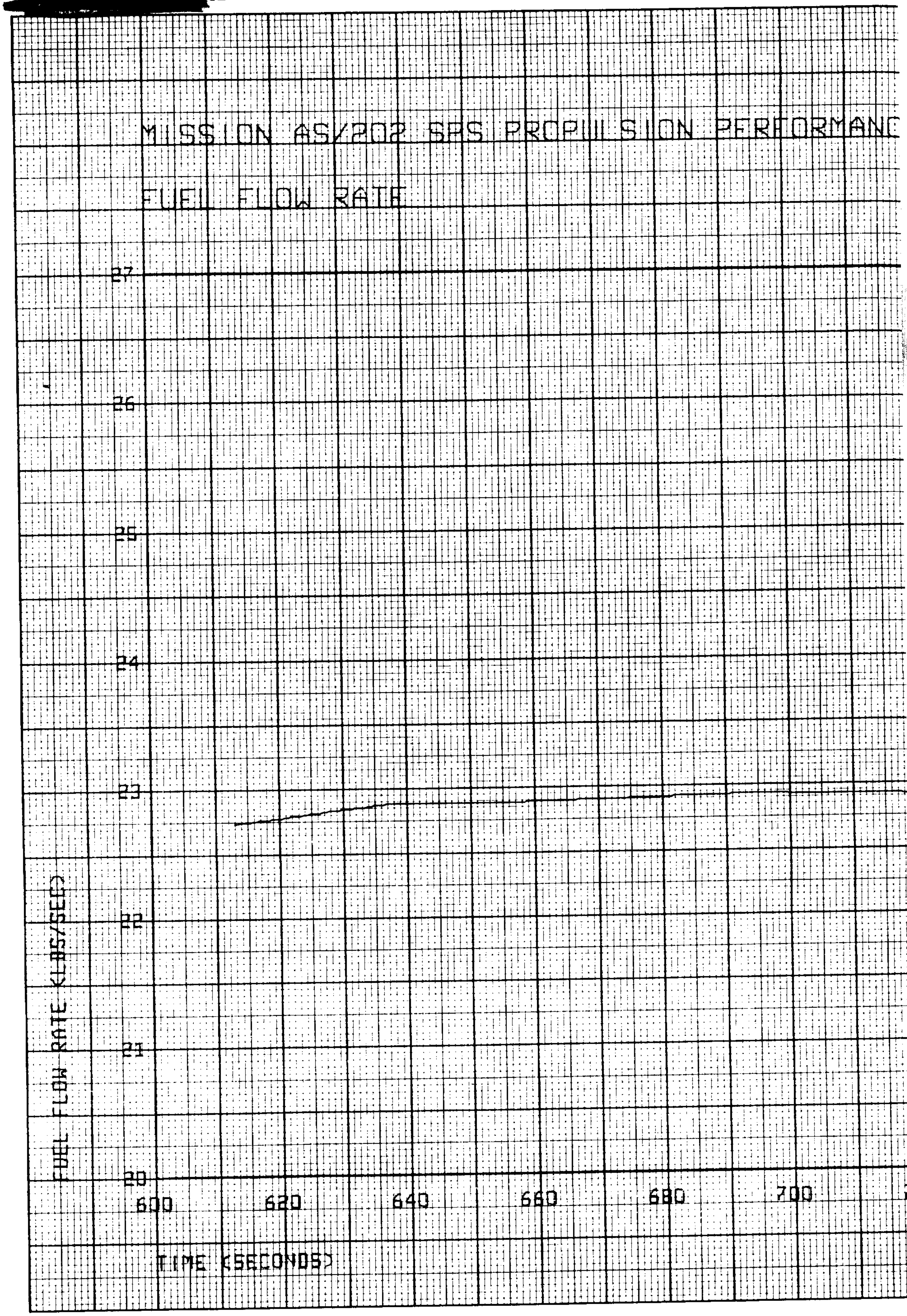

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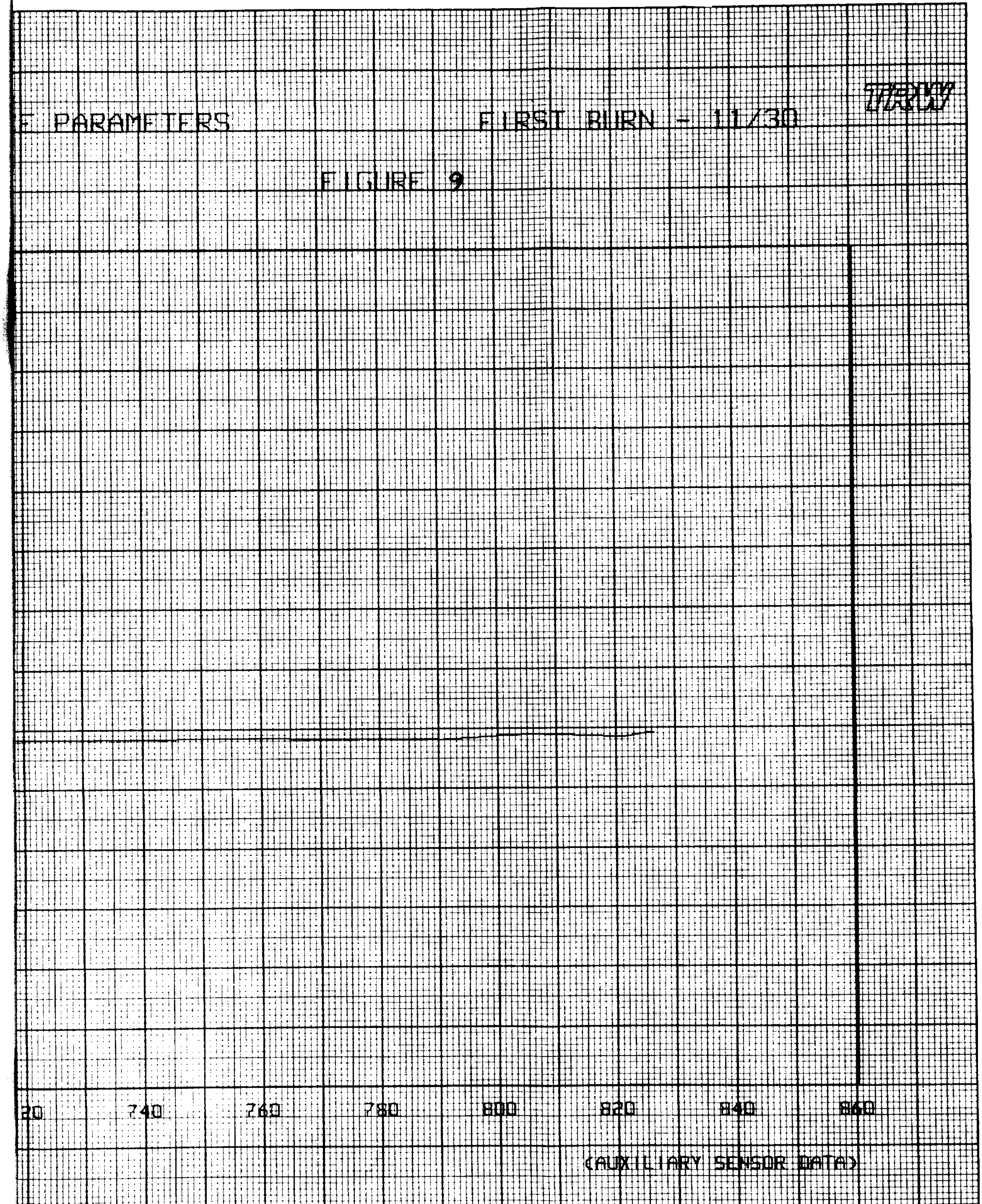

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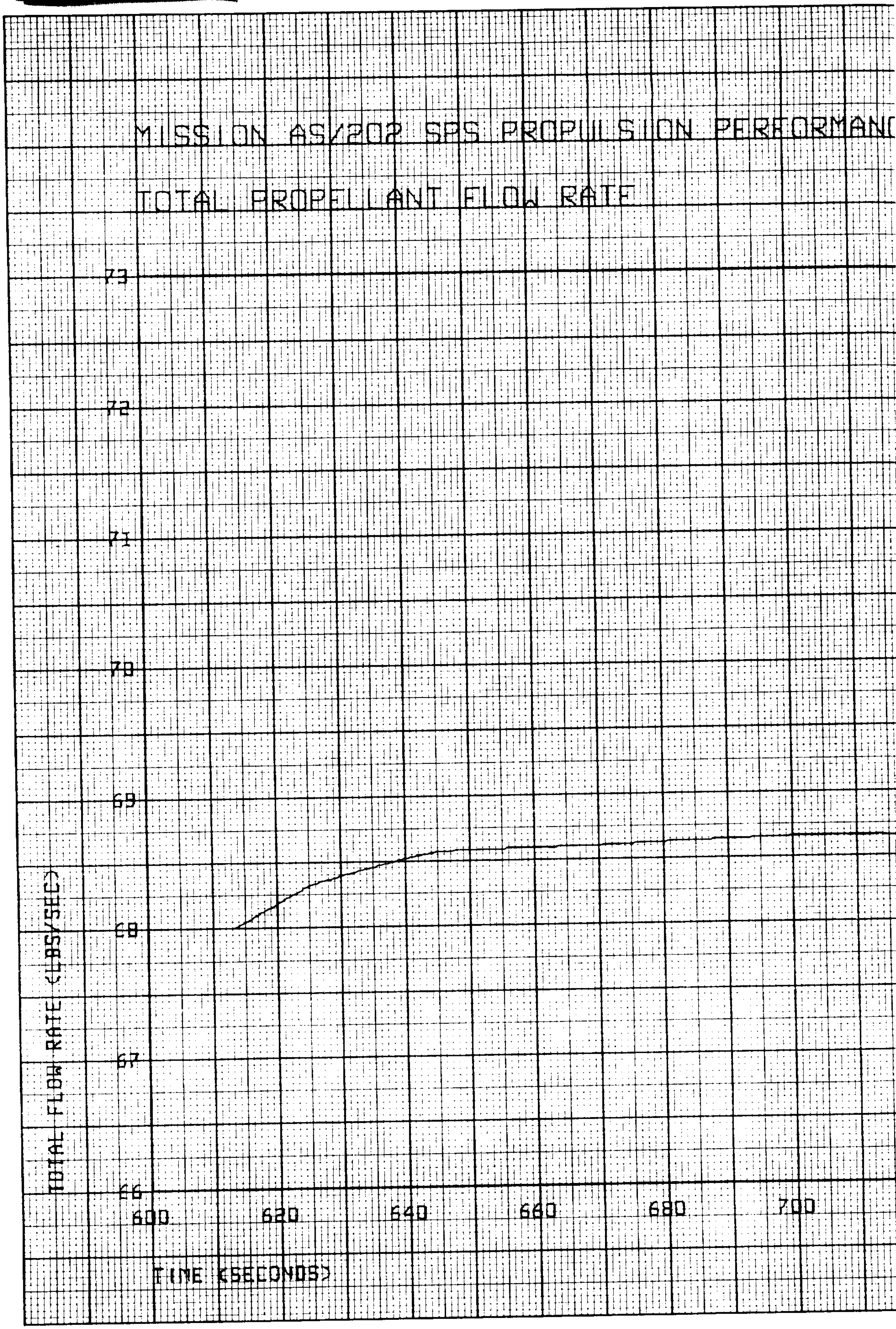

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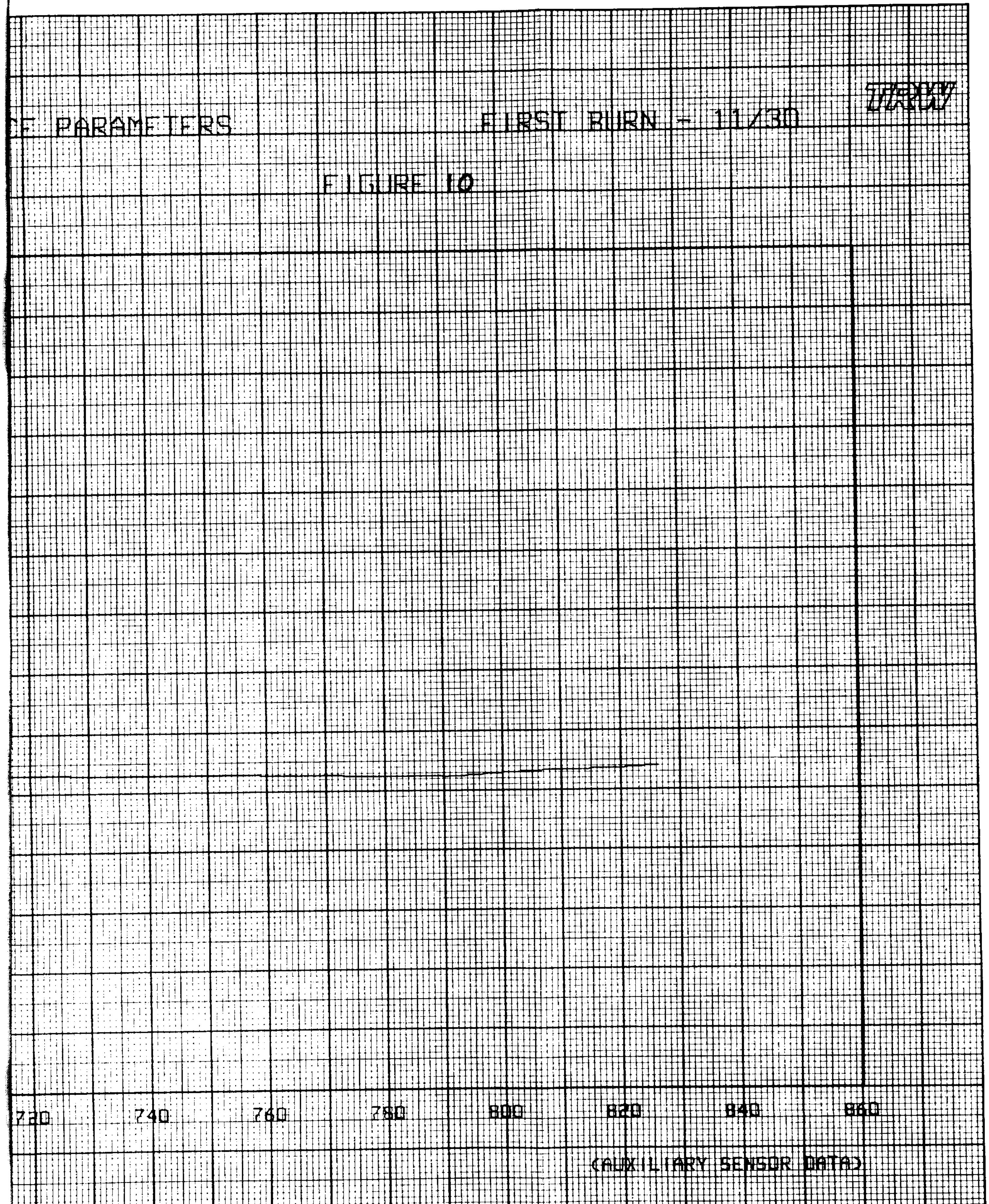

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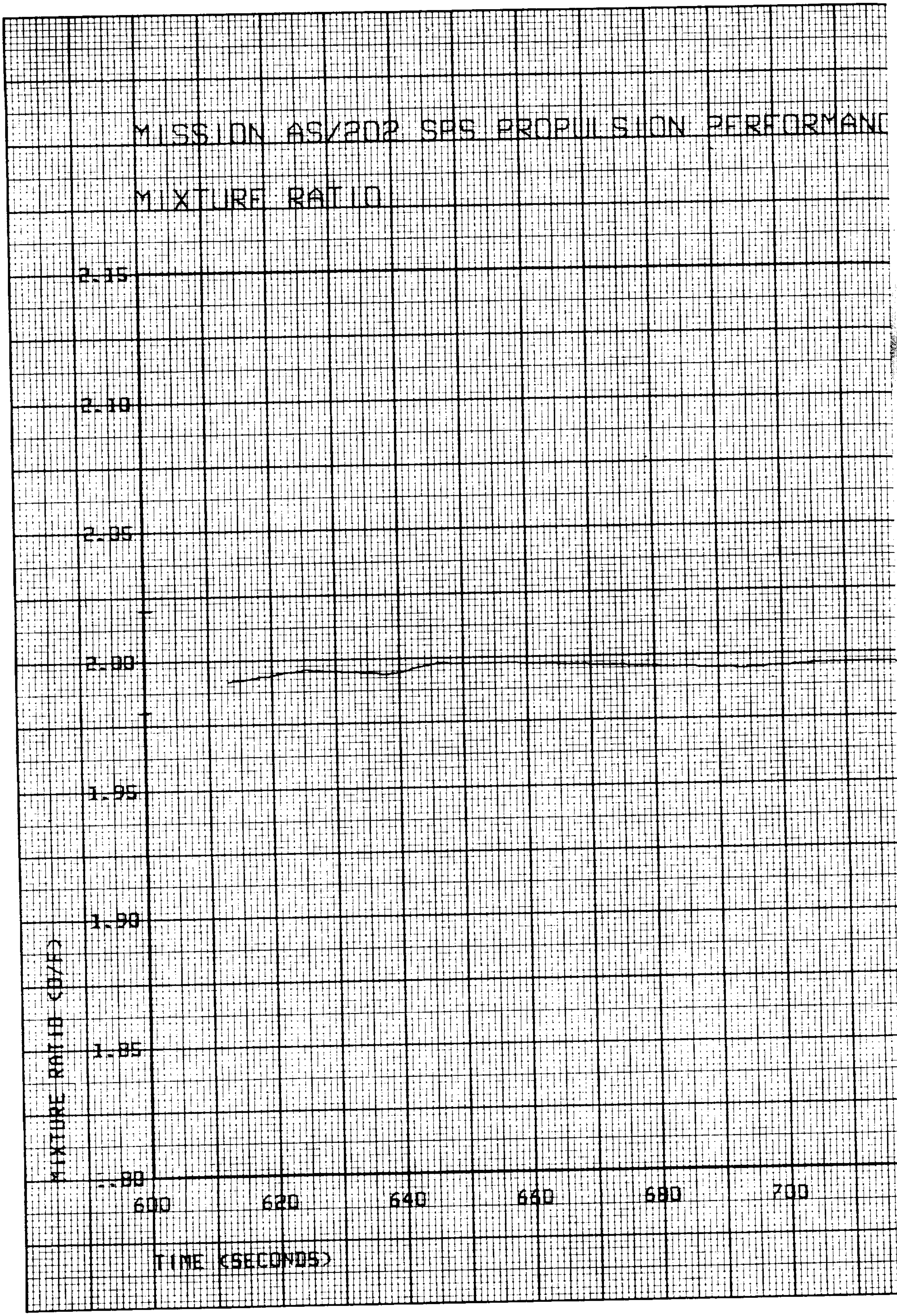

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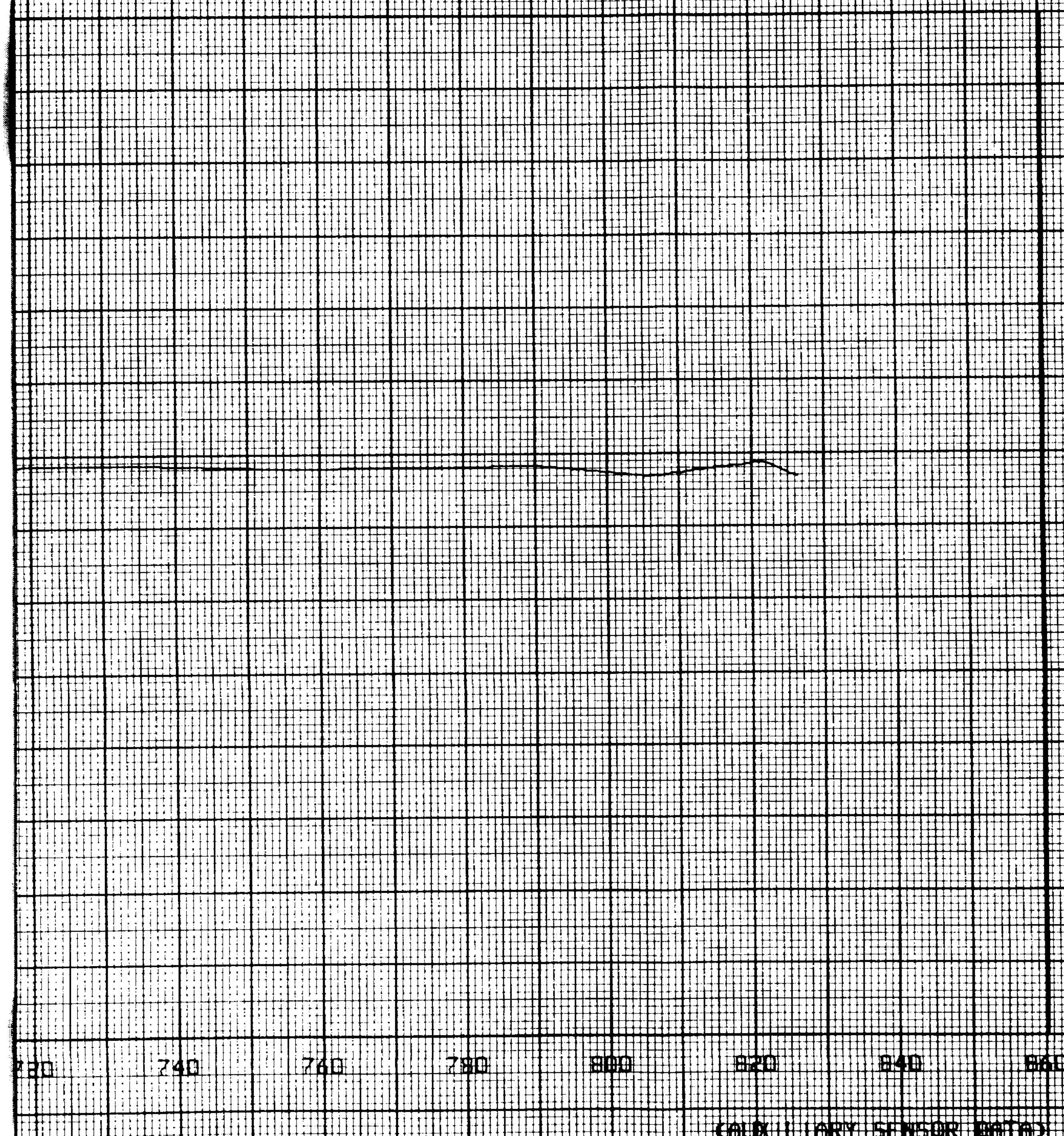
E PARAMETERS FIRST BURN - 11/30

FIGURE 11



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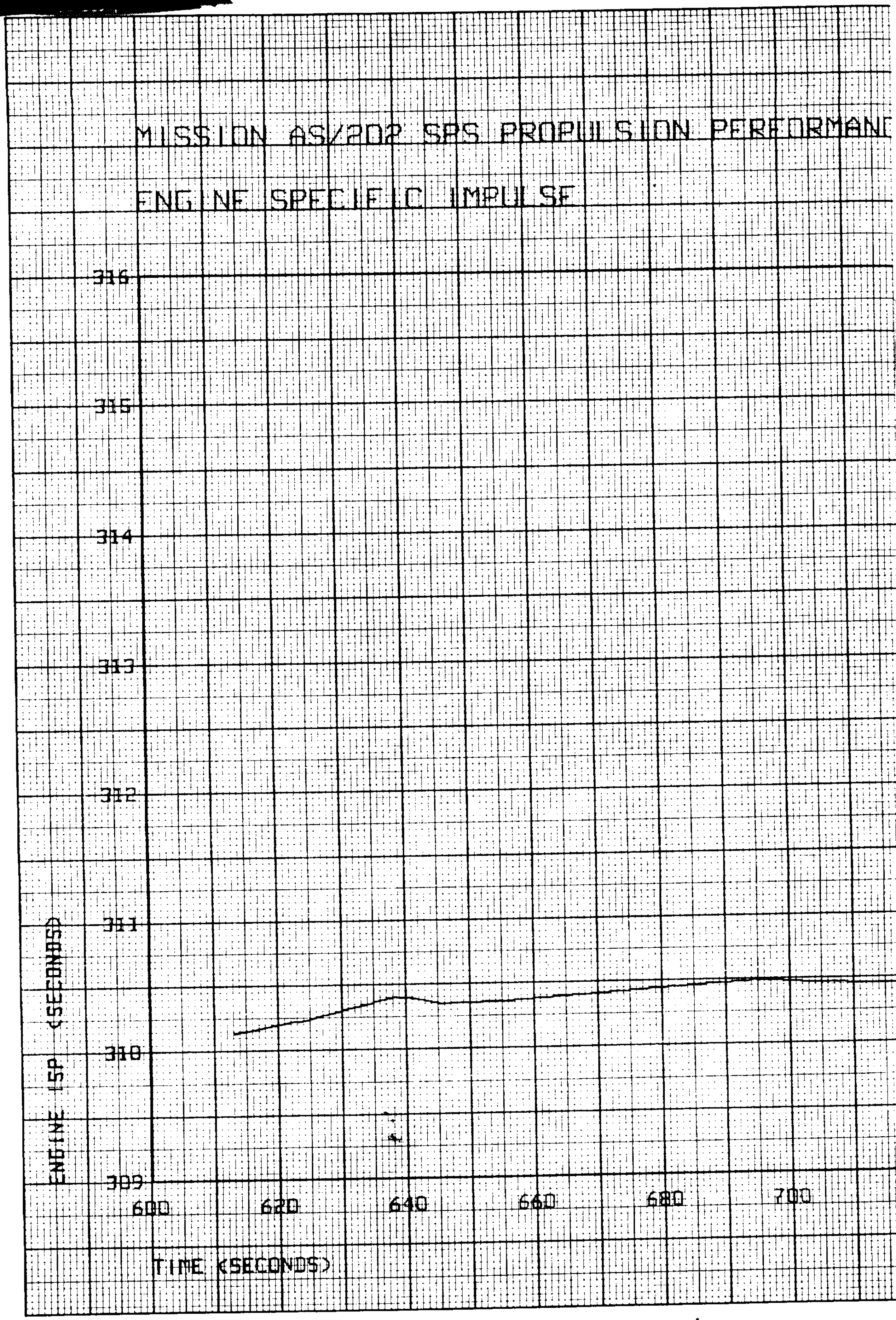

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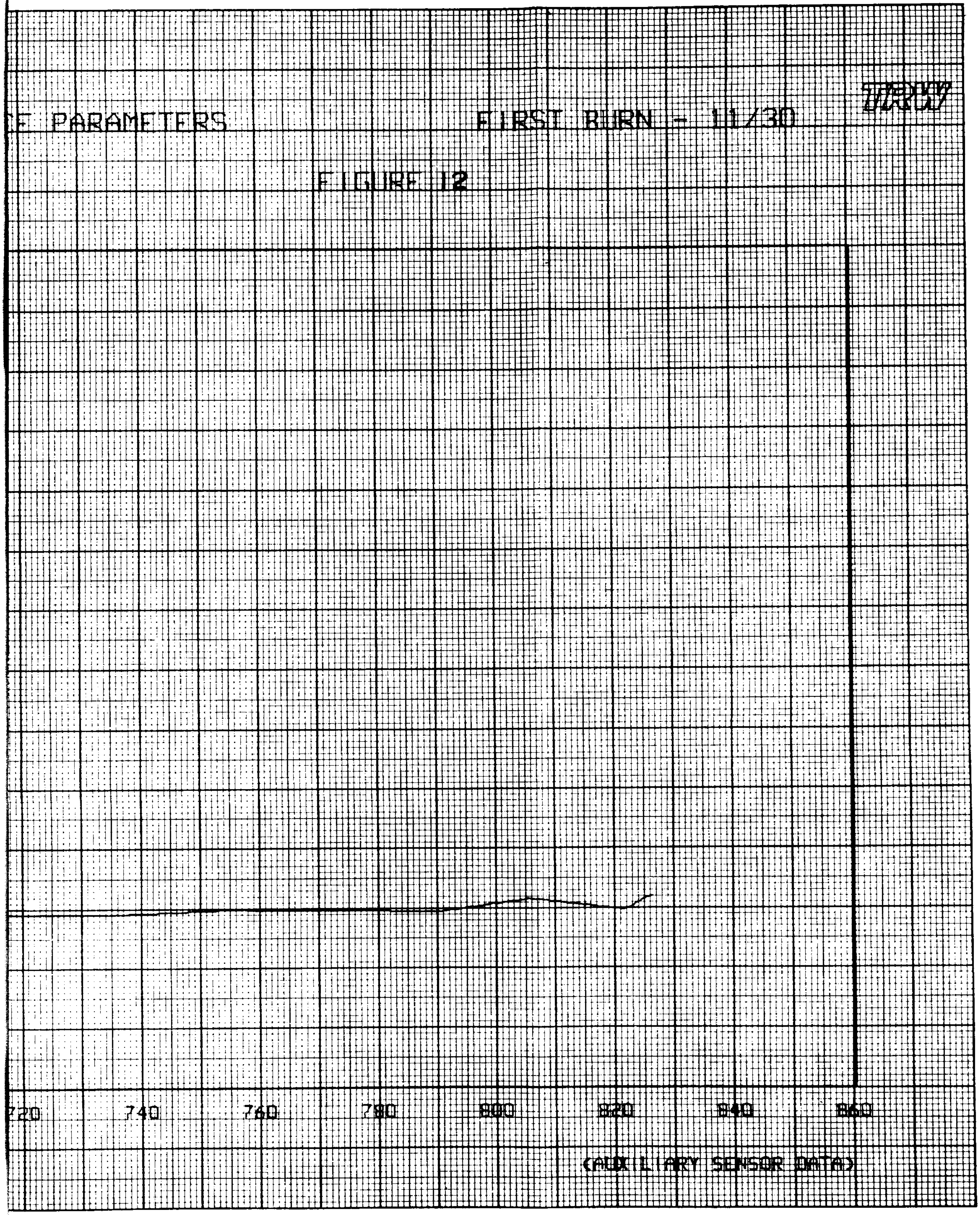
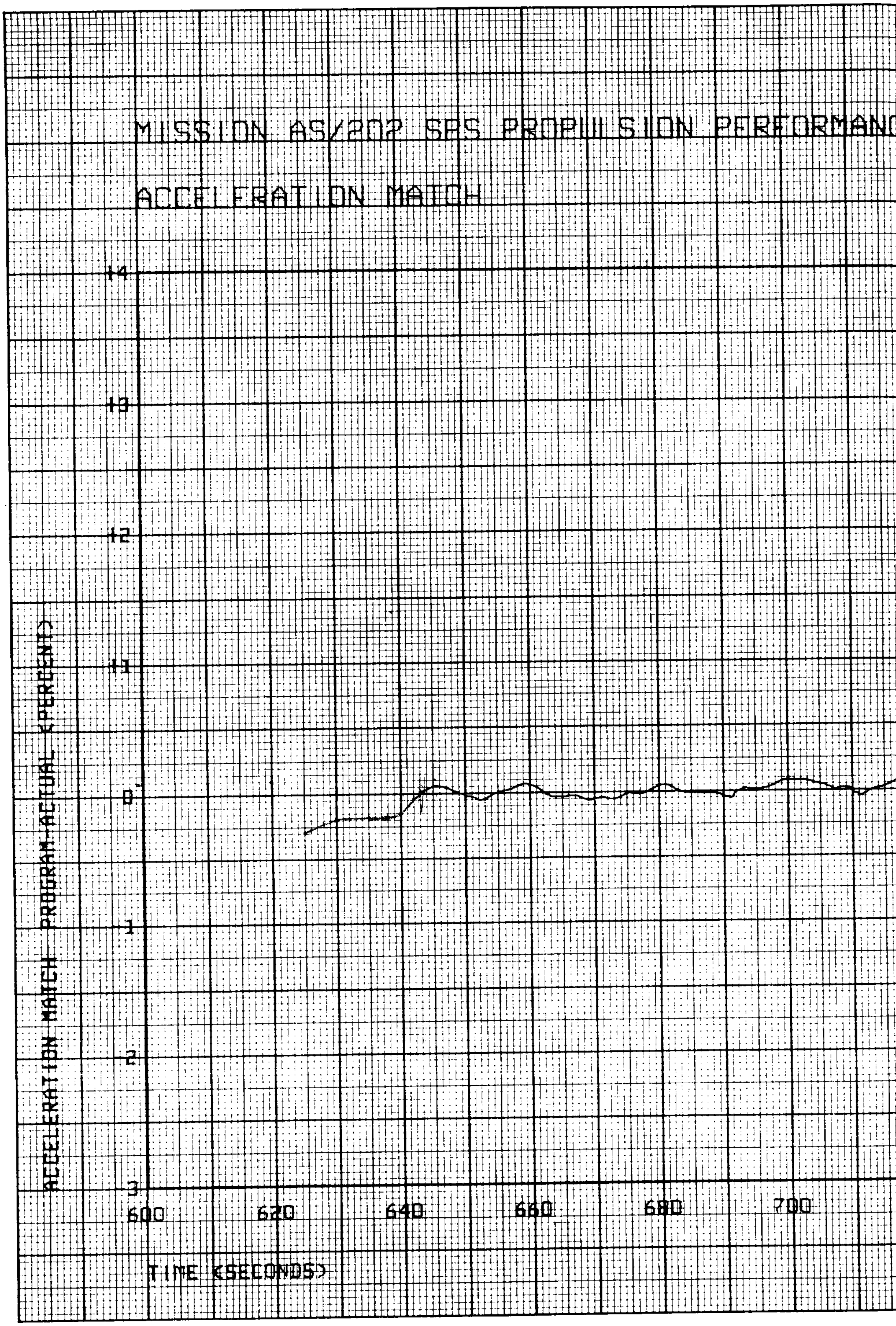

E PARAMETERS
FIRST BURN - 11/30
FIGURE 12
720
740
760
780
800
820
840
860
(AUXILIARY SENSOR DATA)



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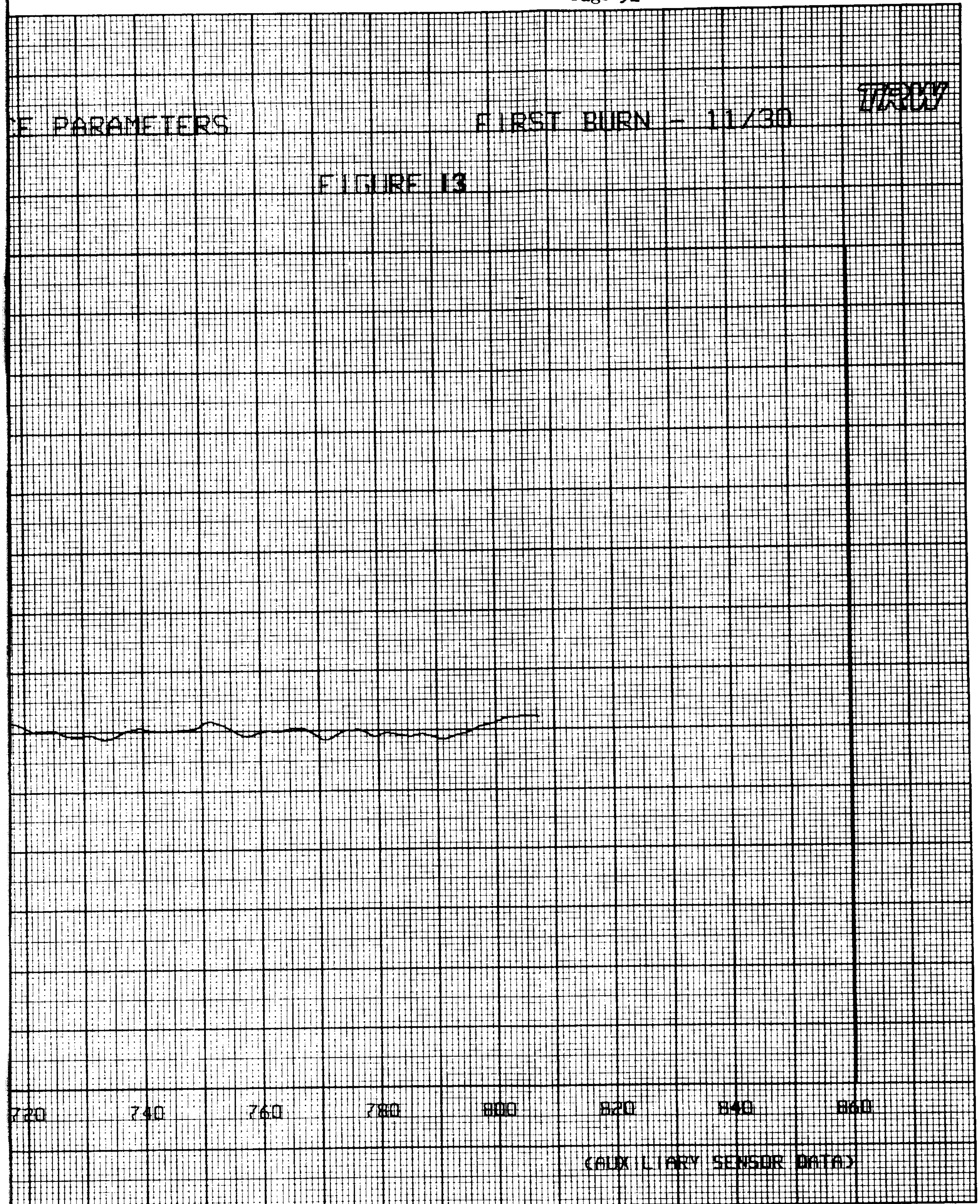

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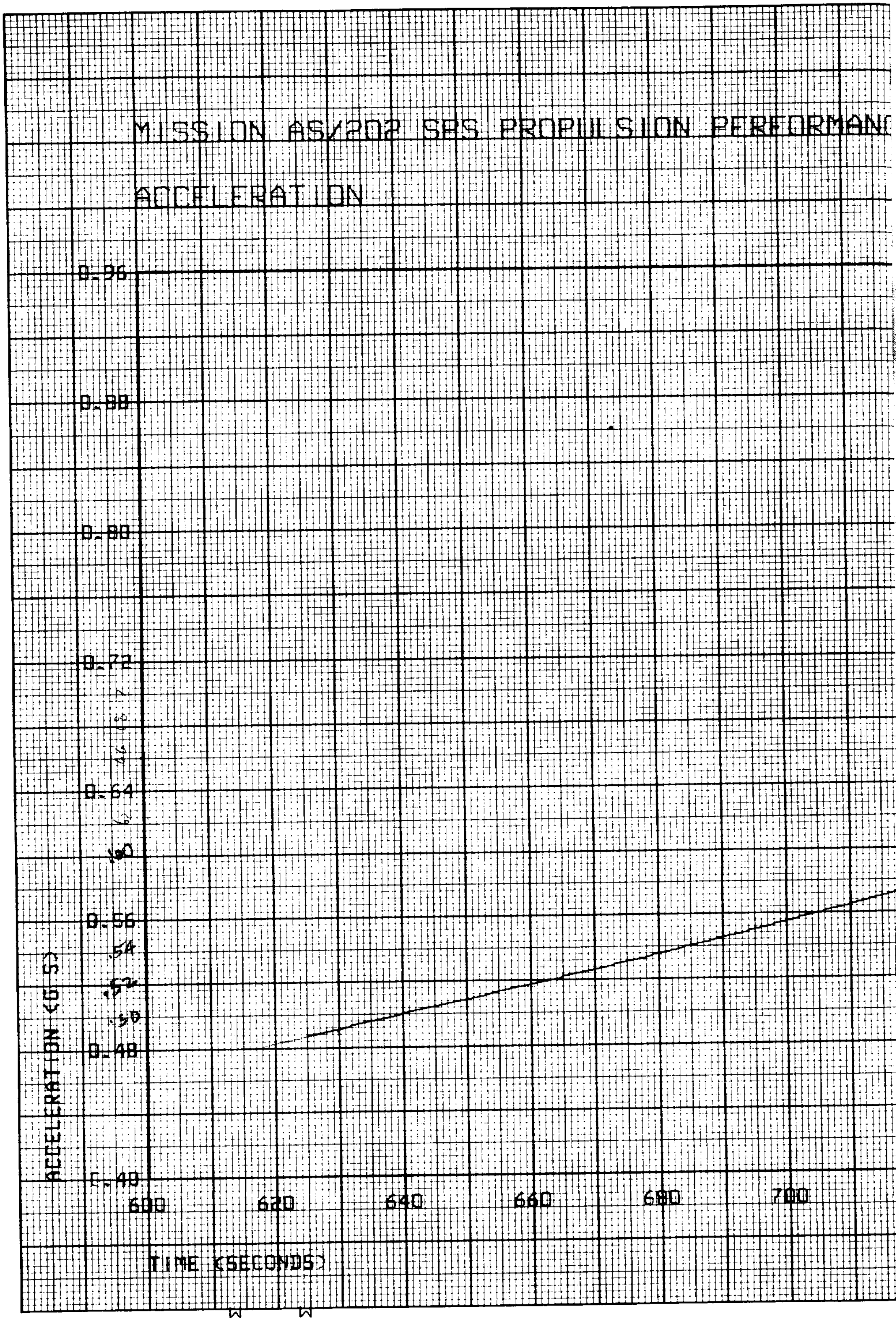

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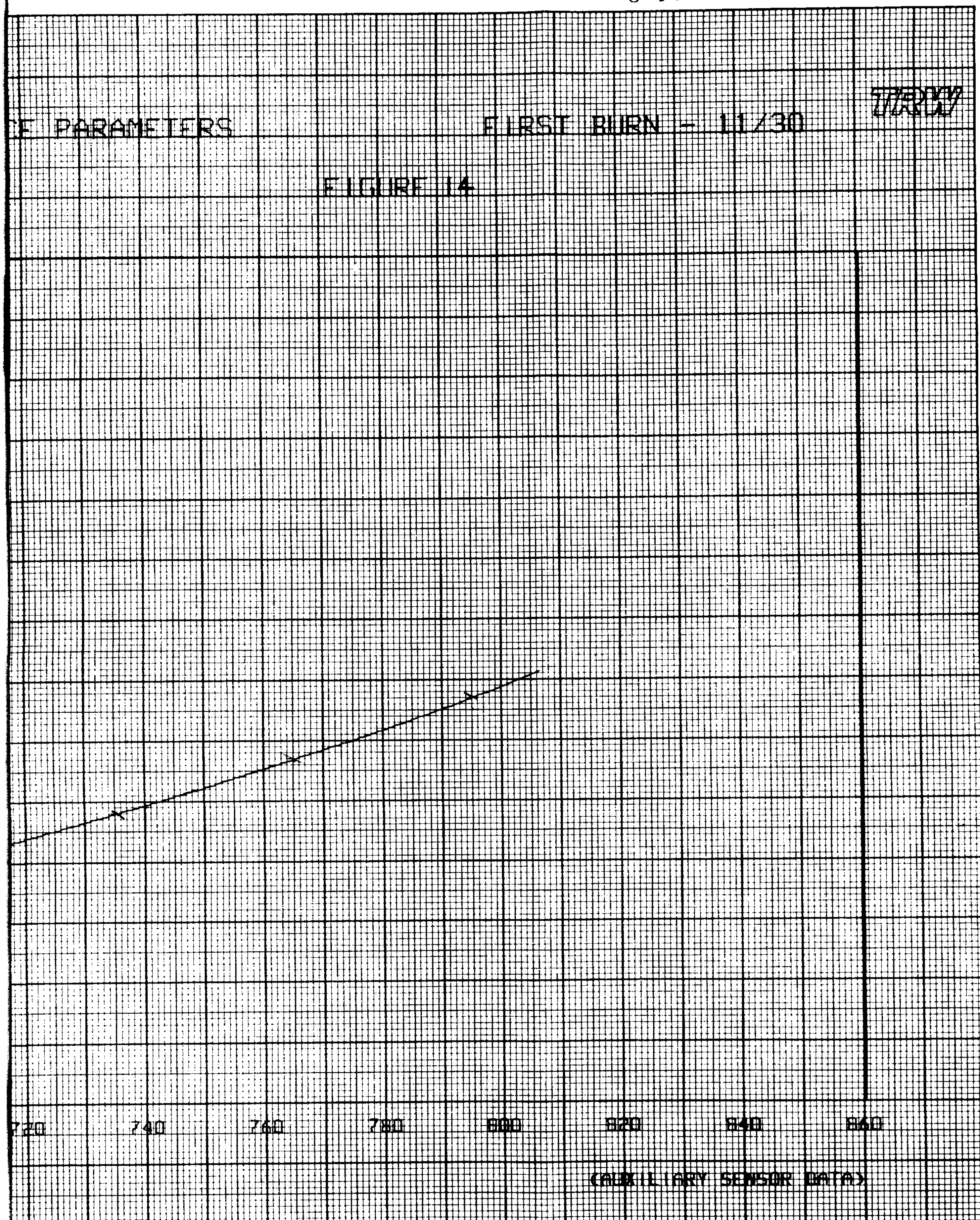

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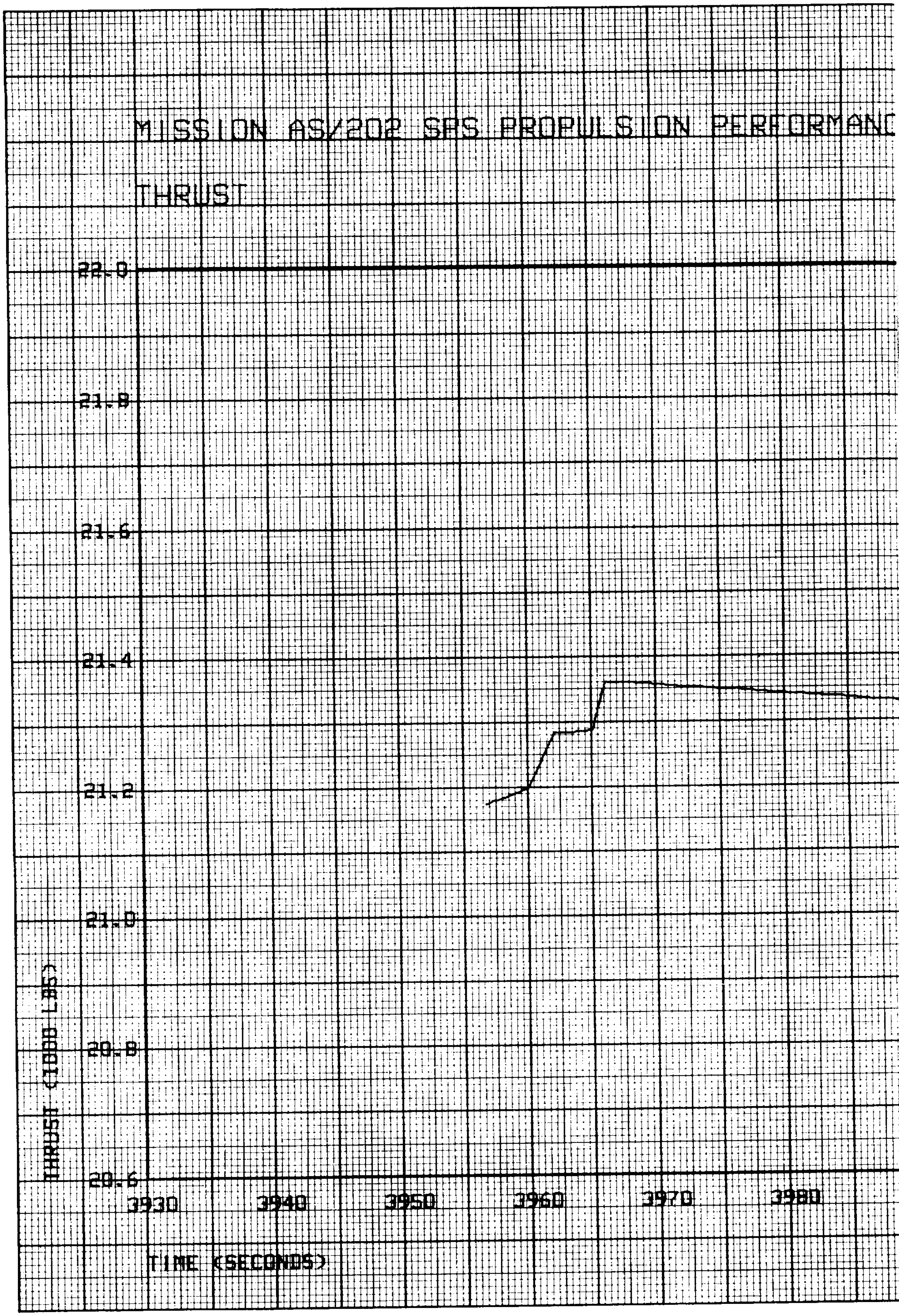

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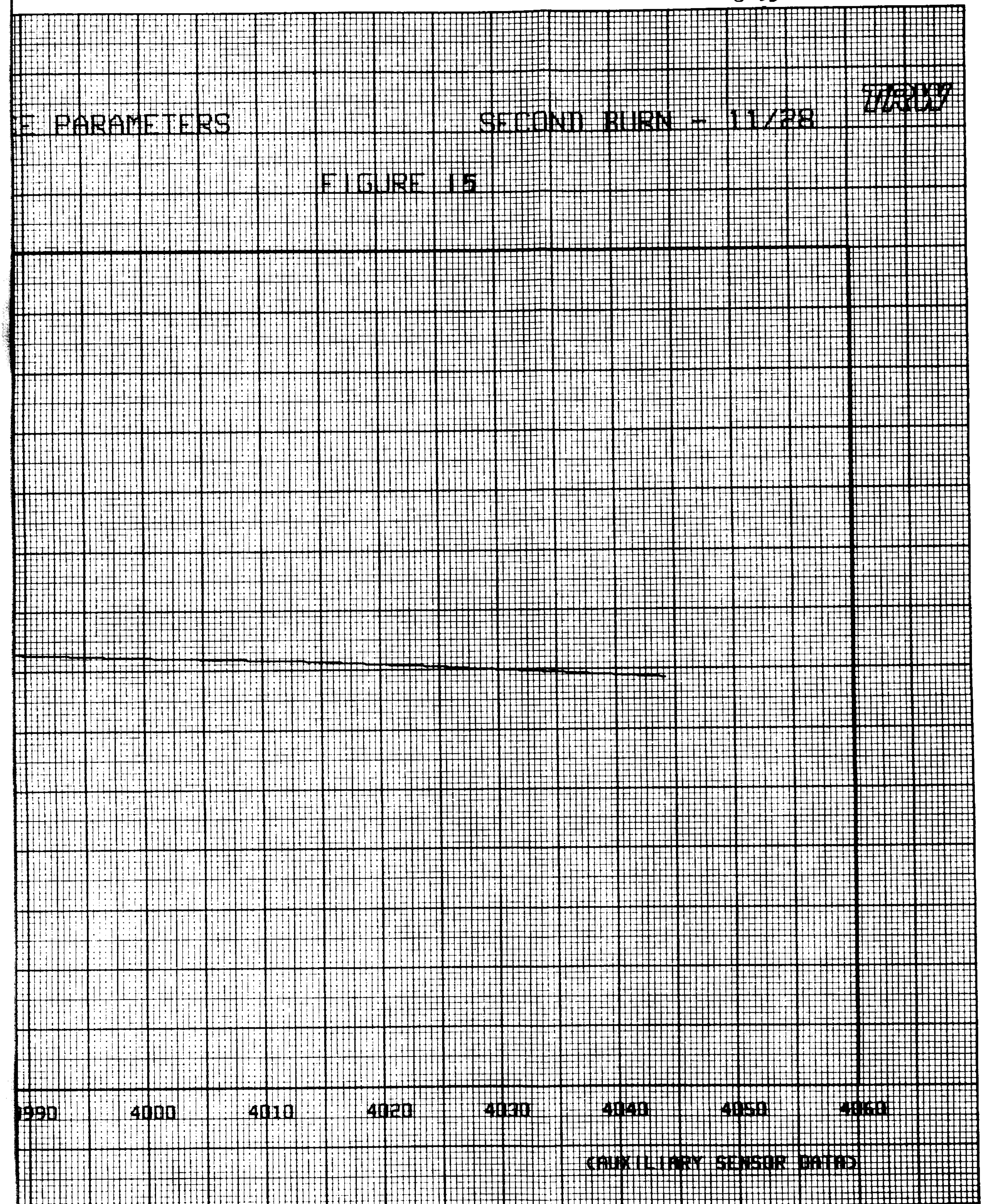

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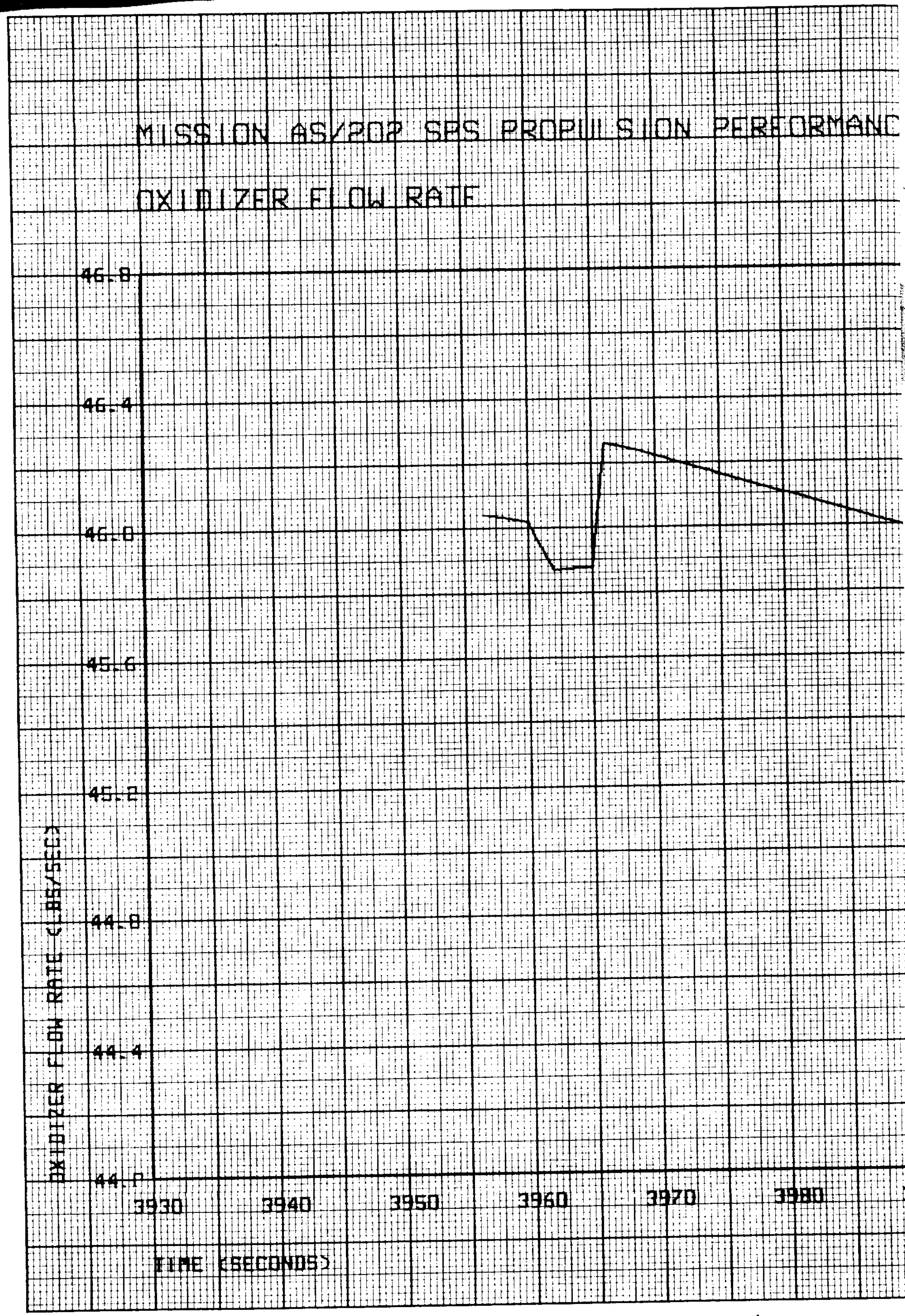

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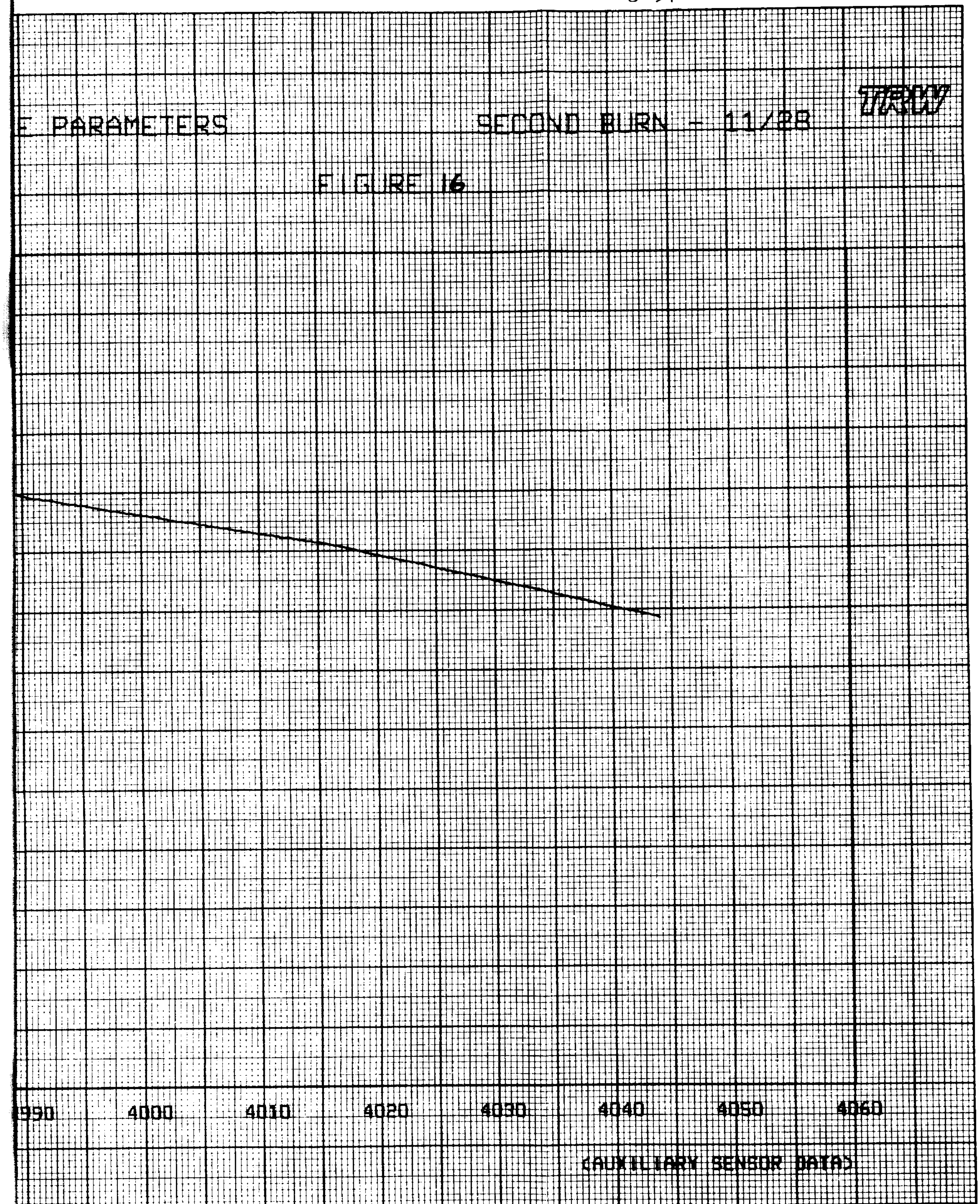

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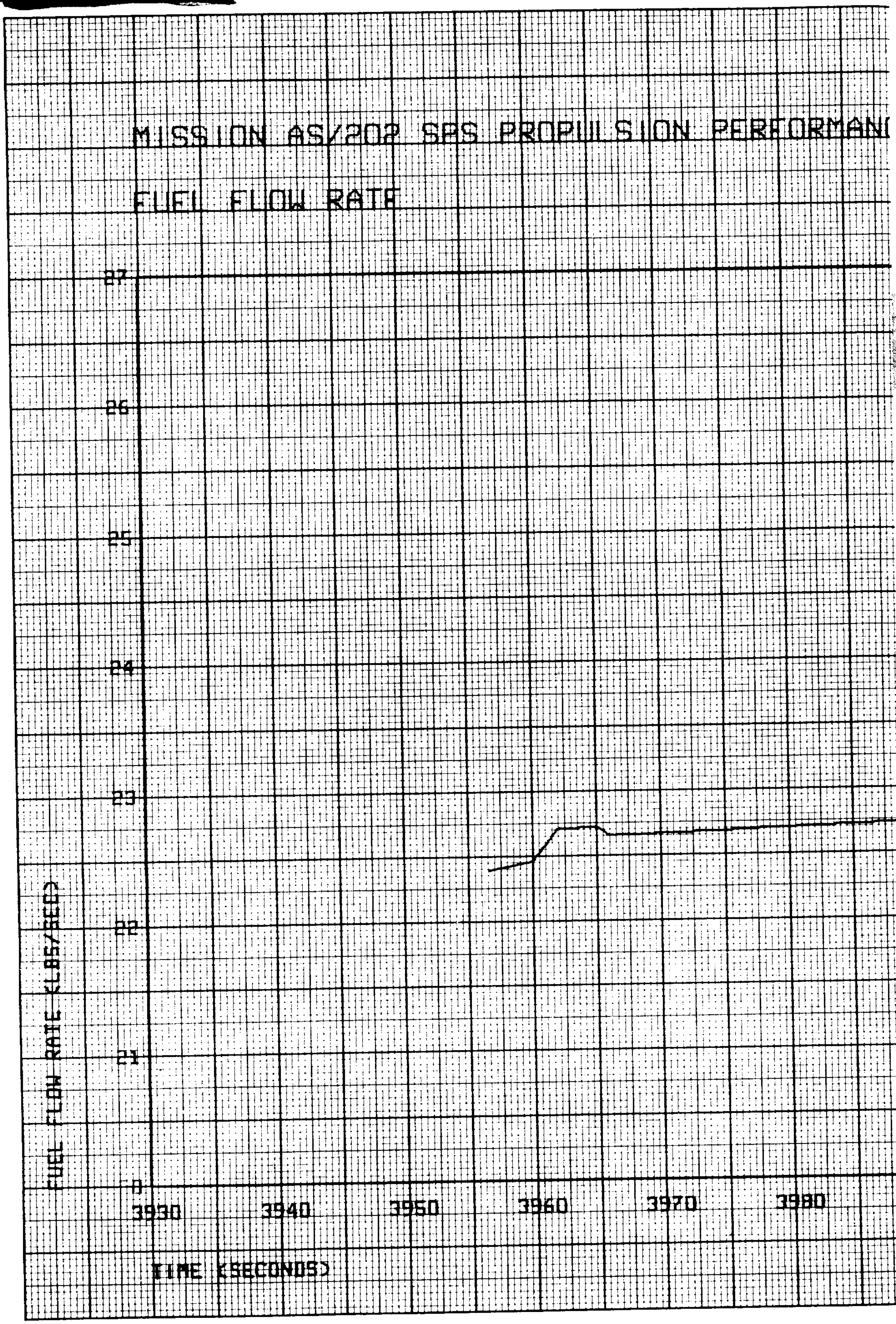

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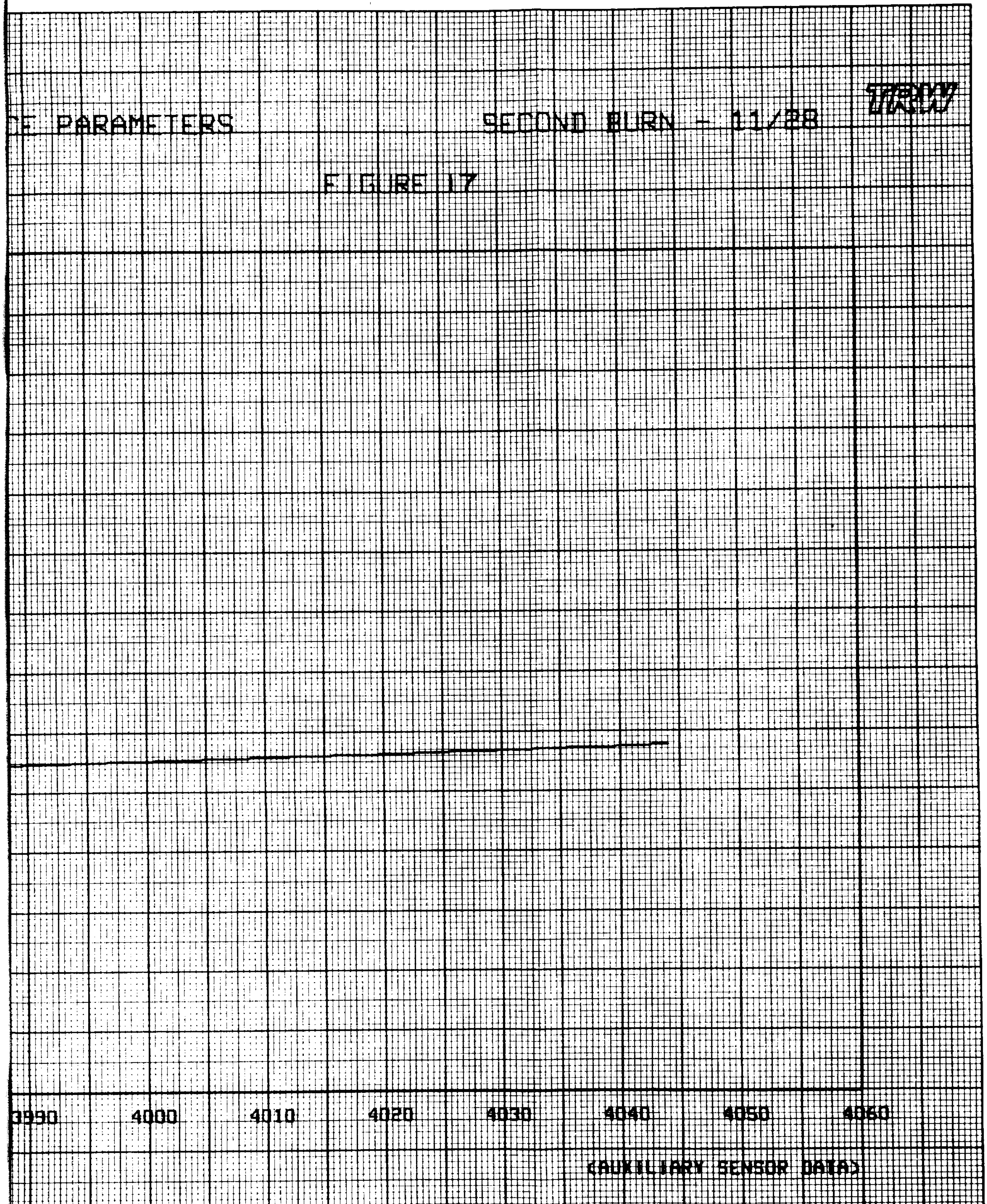

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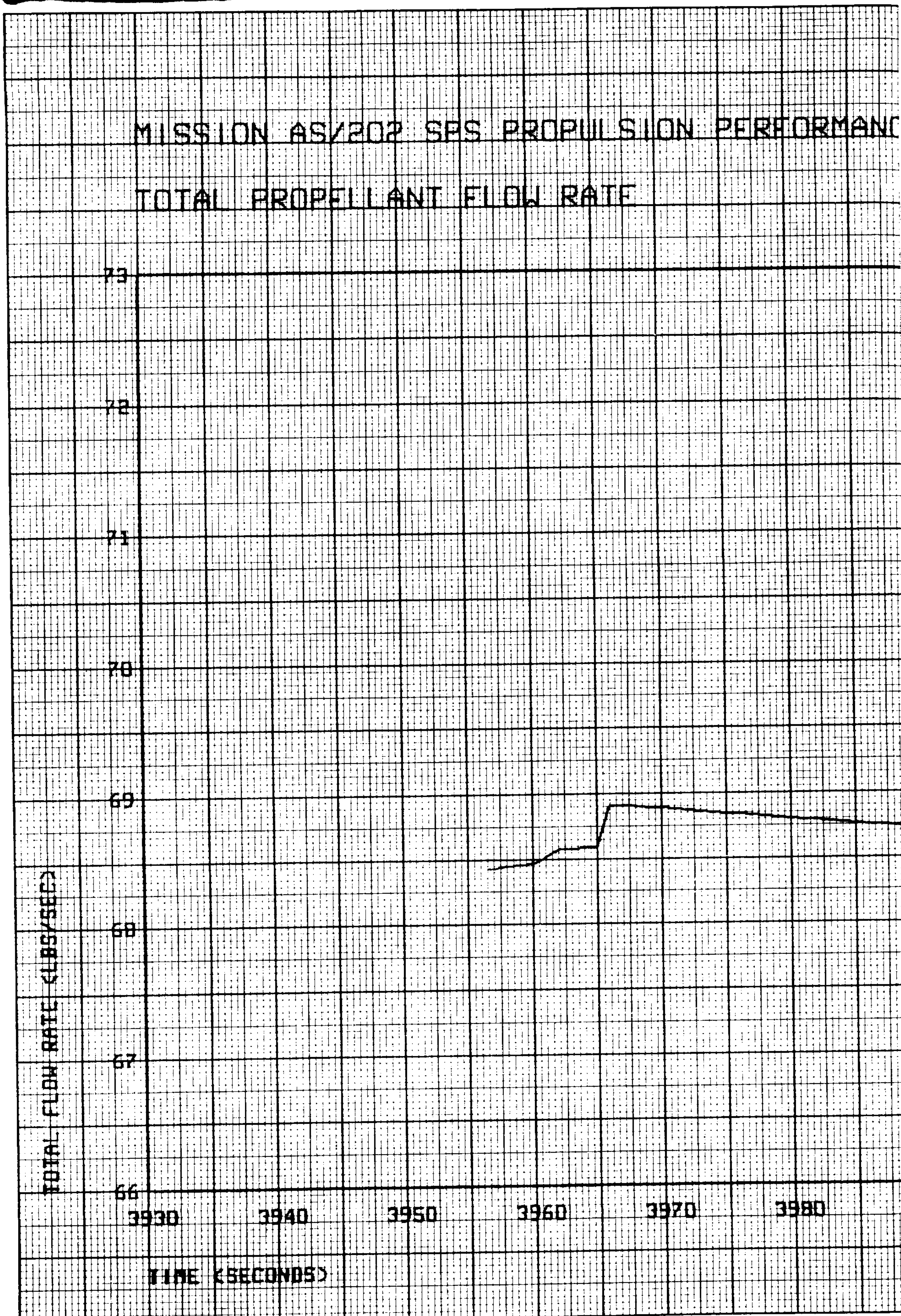

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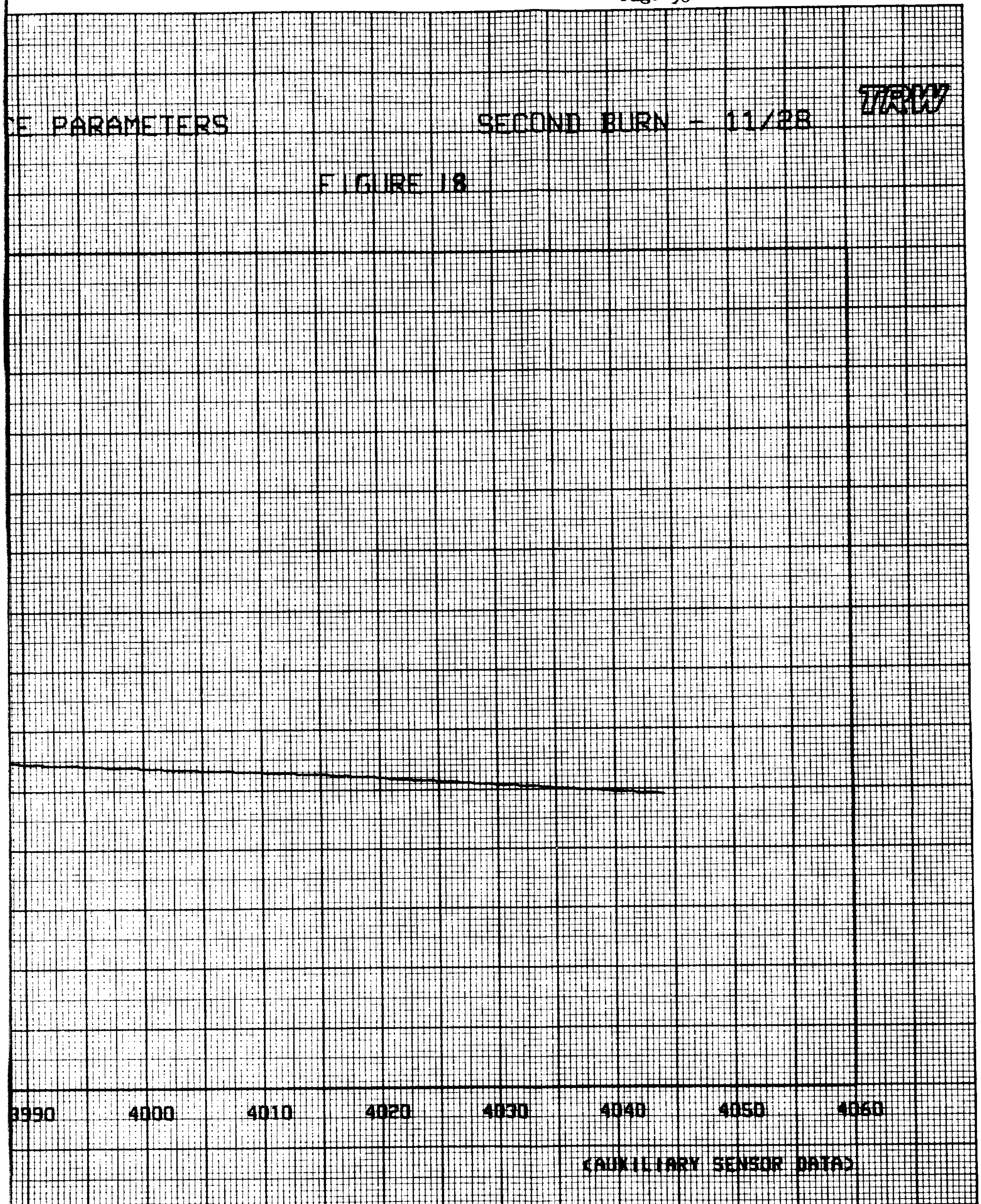

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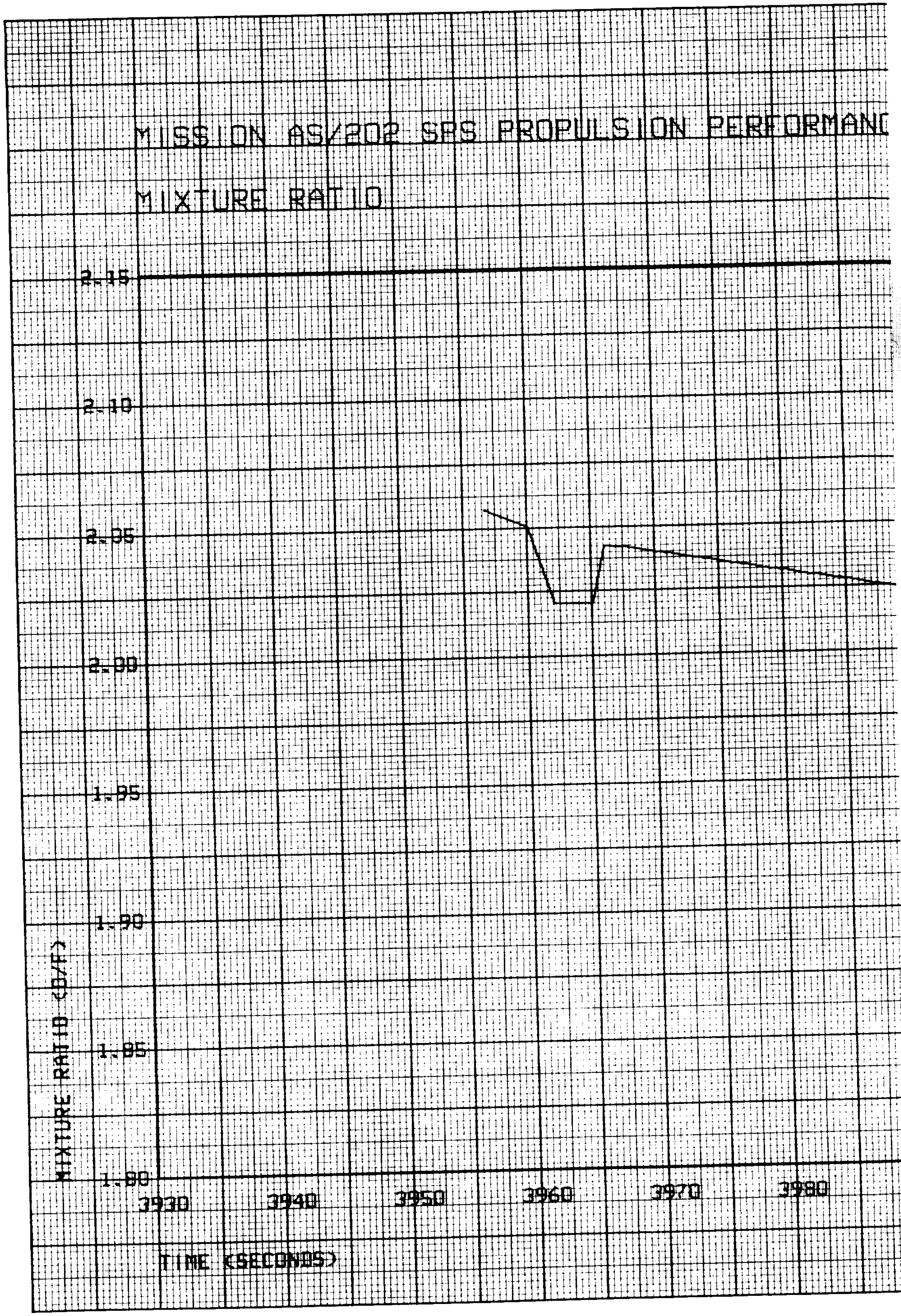

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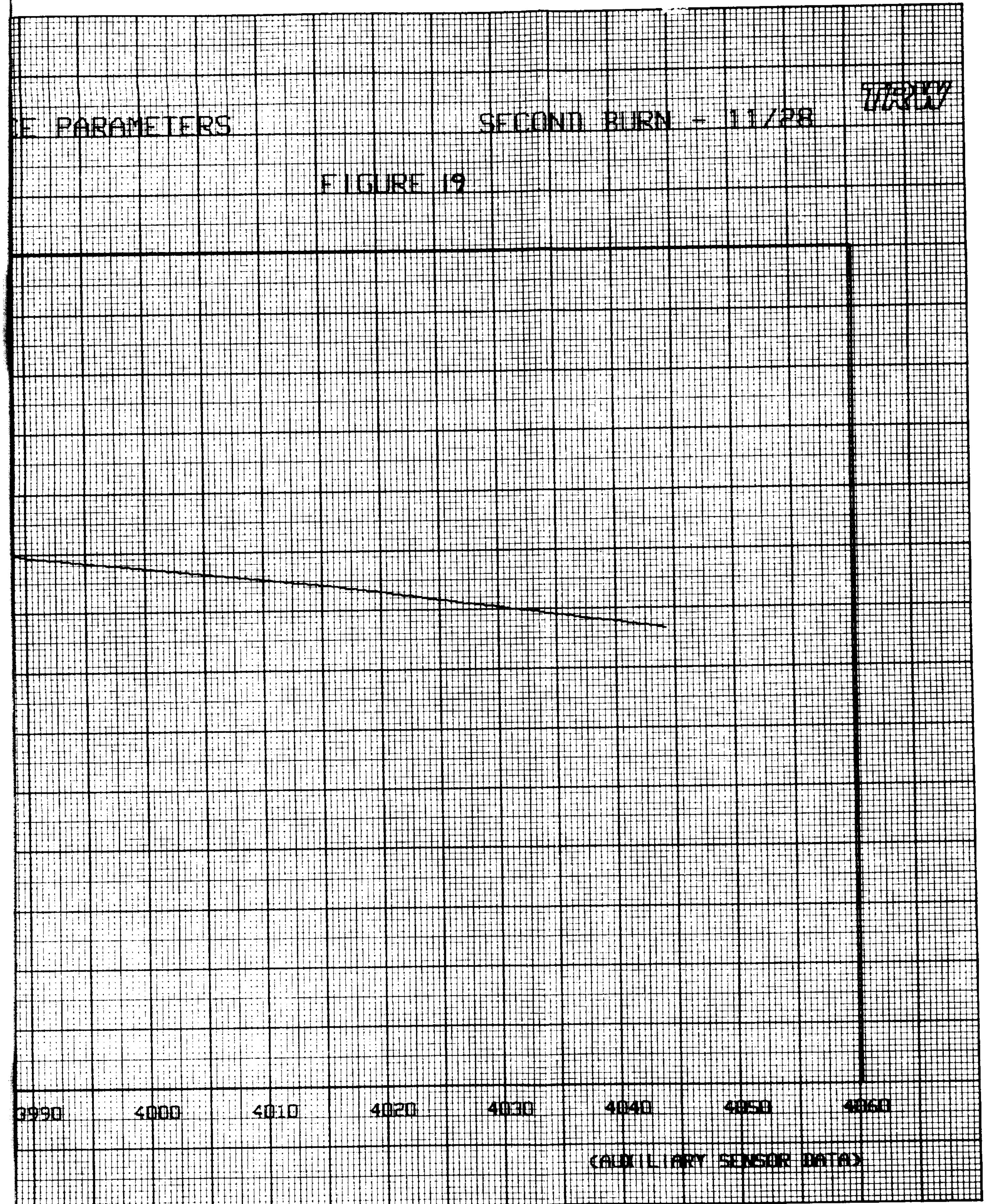
E PARAMETERS
SECOND BURN - 11/28
FIGURE 19
3990
4000
4010
4020
4030
4040
4050
4060
(AUXILIARY SENSOR DATA)

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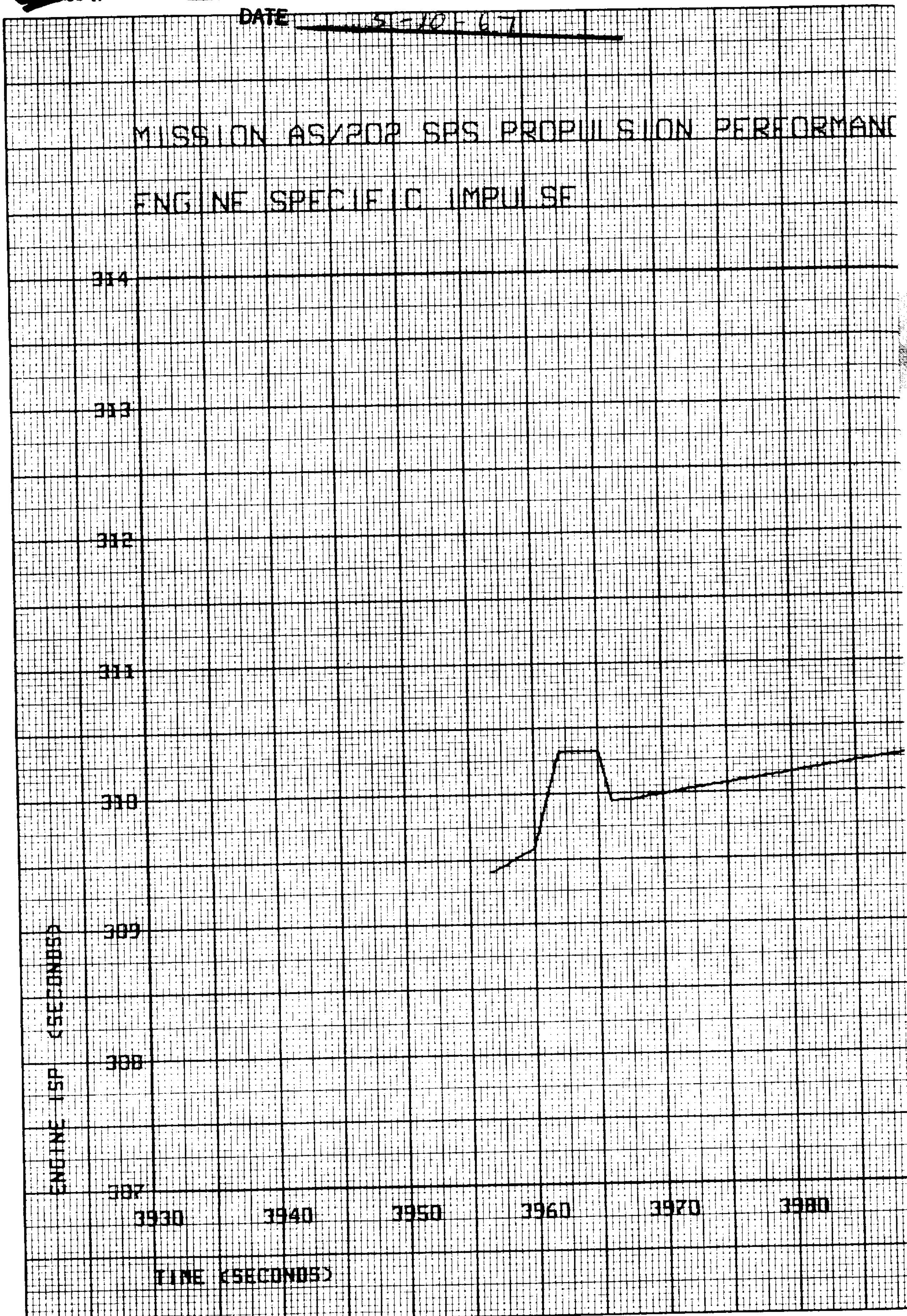
CLASSIFIED CHANGED TO UNCLASSIFIED
AUTHORITY: SCG - 11 REV. 1 Supp 2-10-67

DATE 5-10-67



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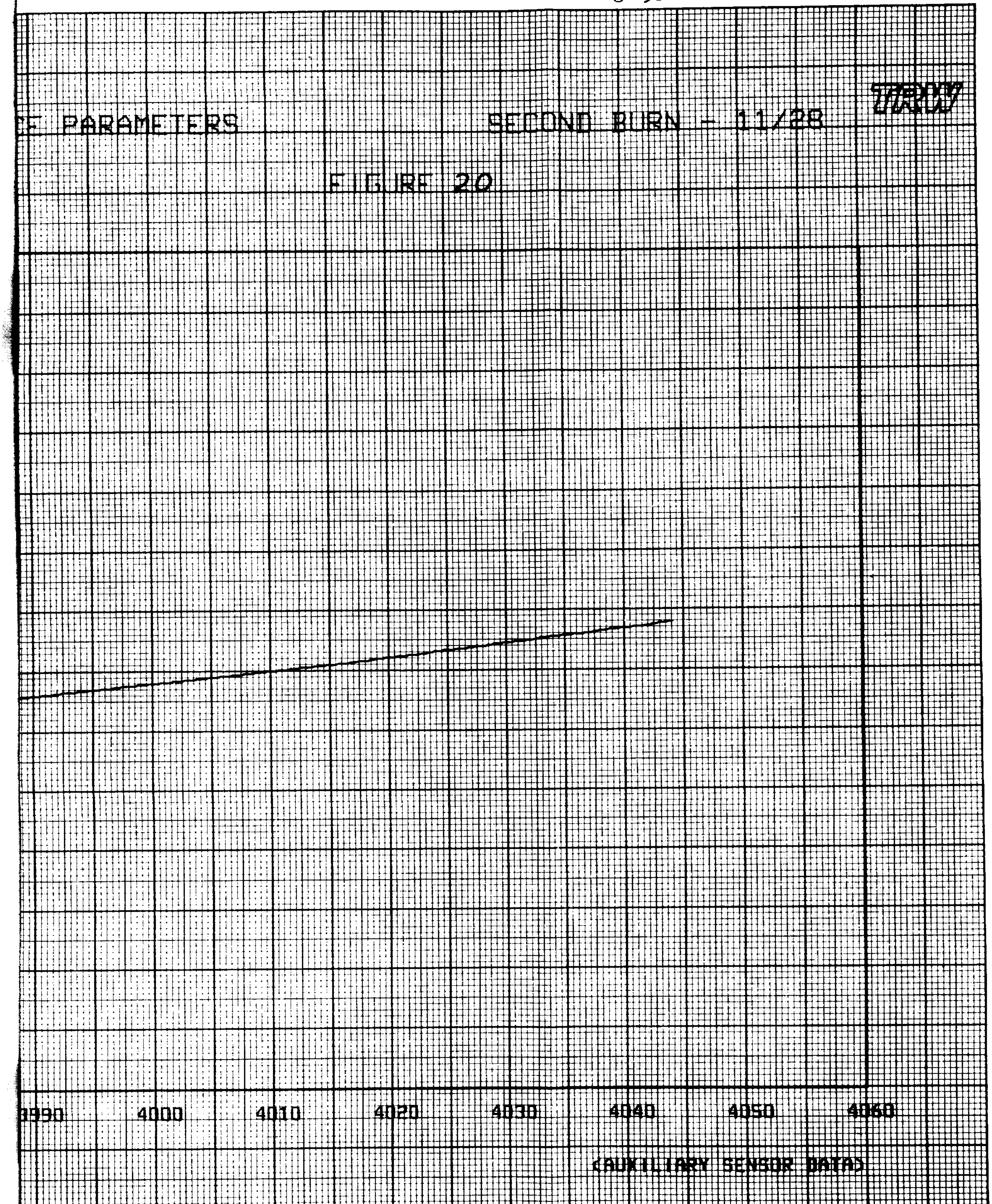
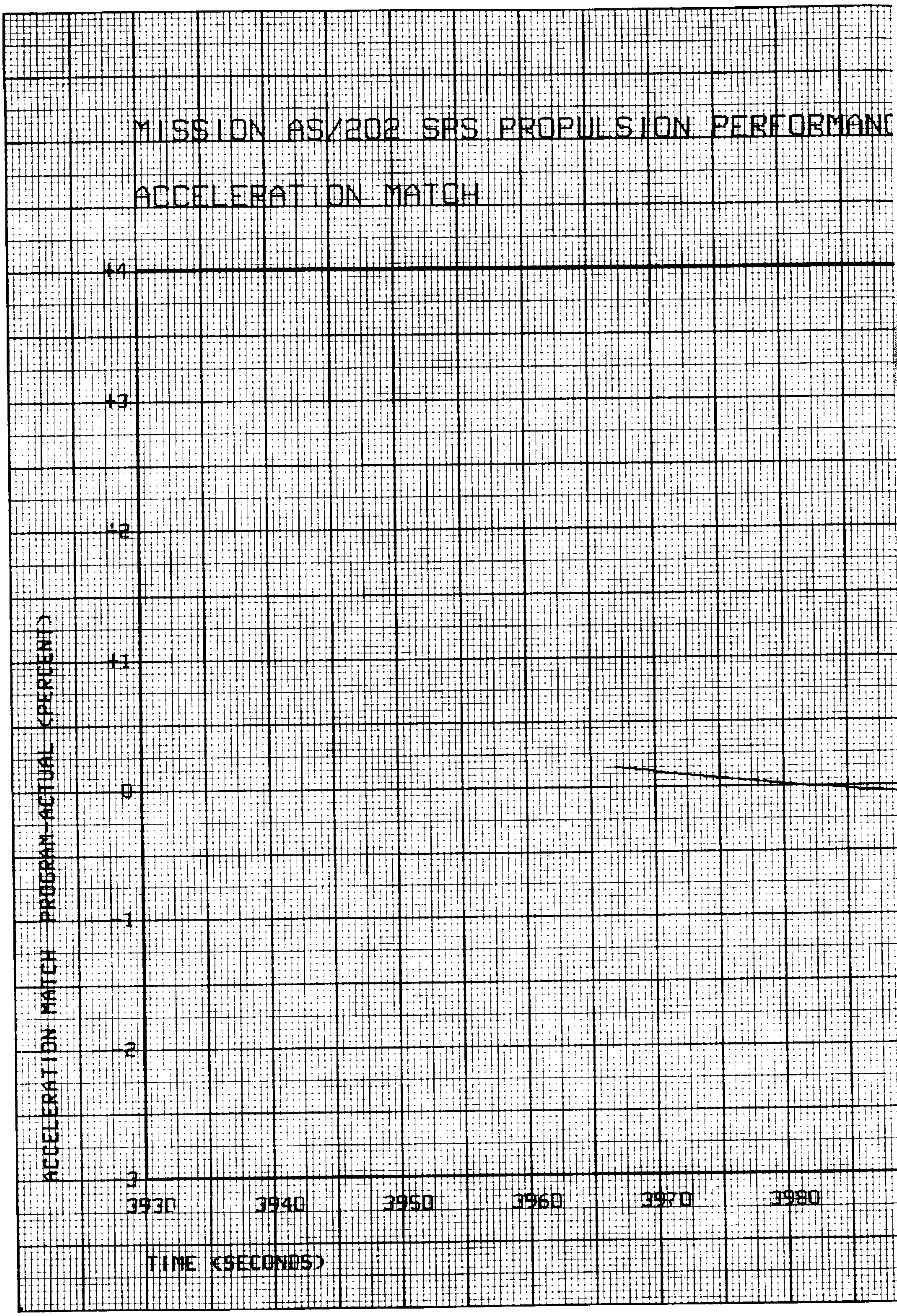
FOLDOUT FRAME 1



FOLDOUT FRAME 2



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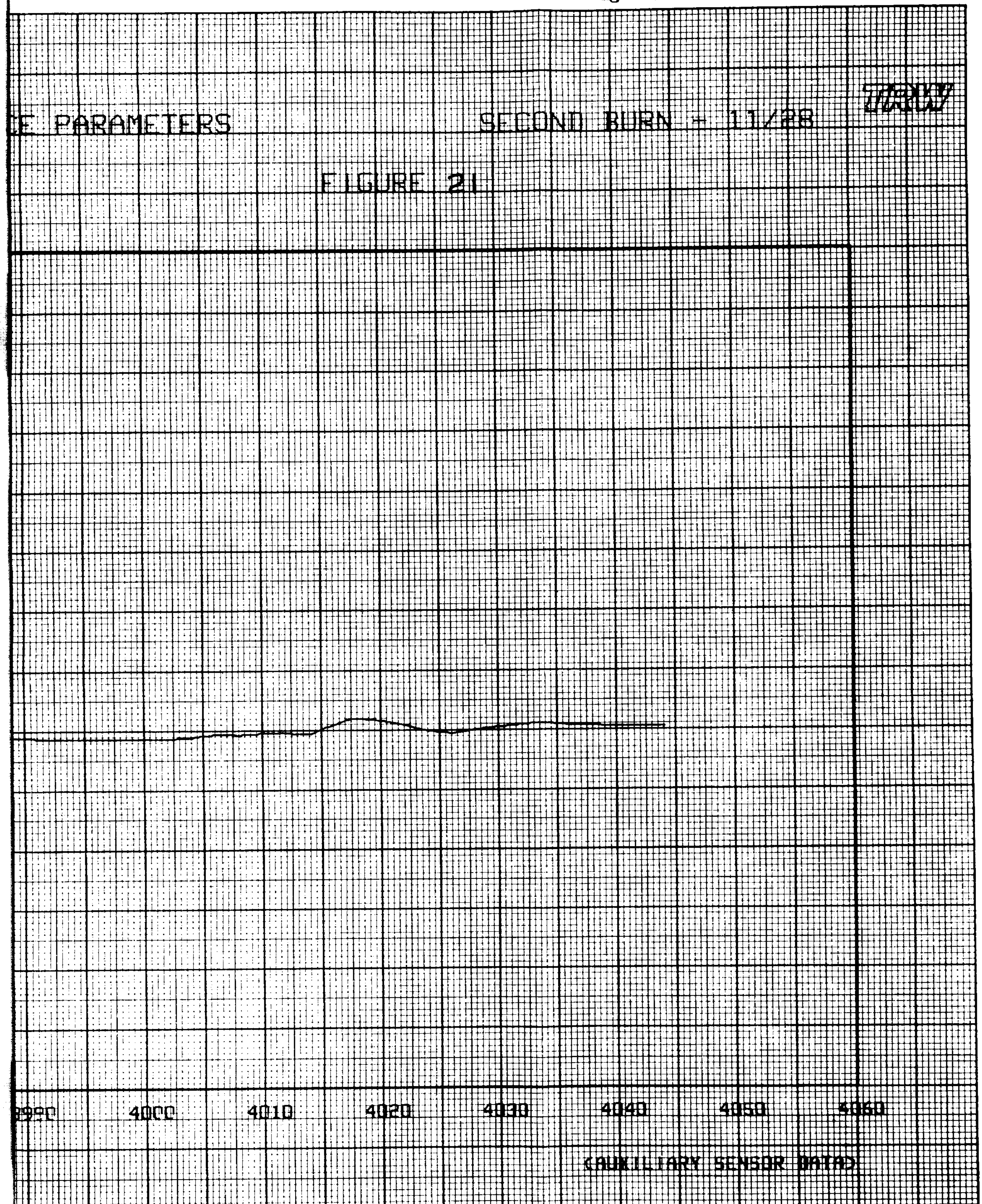

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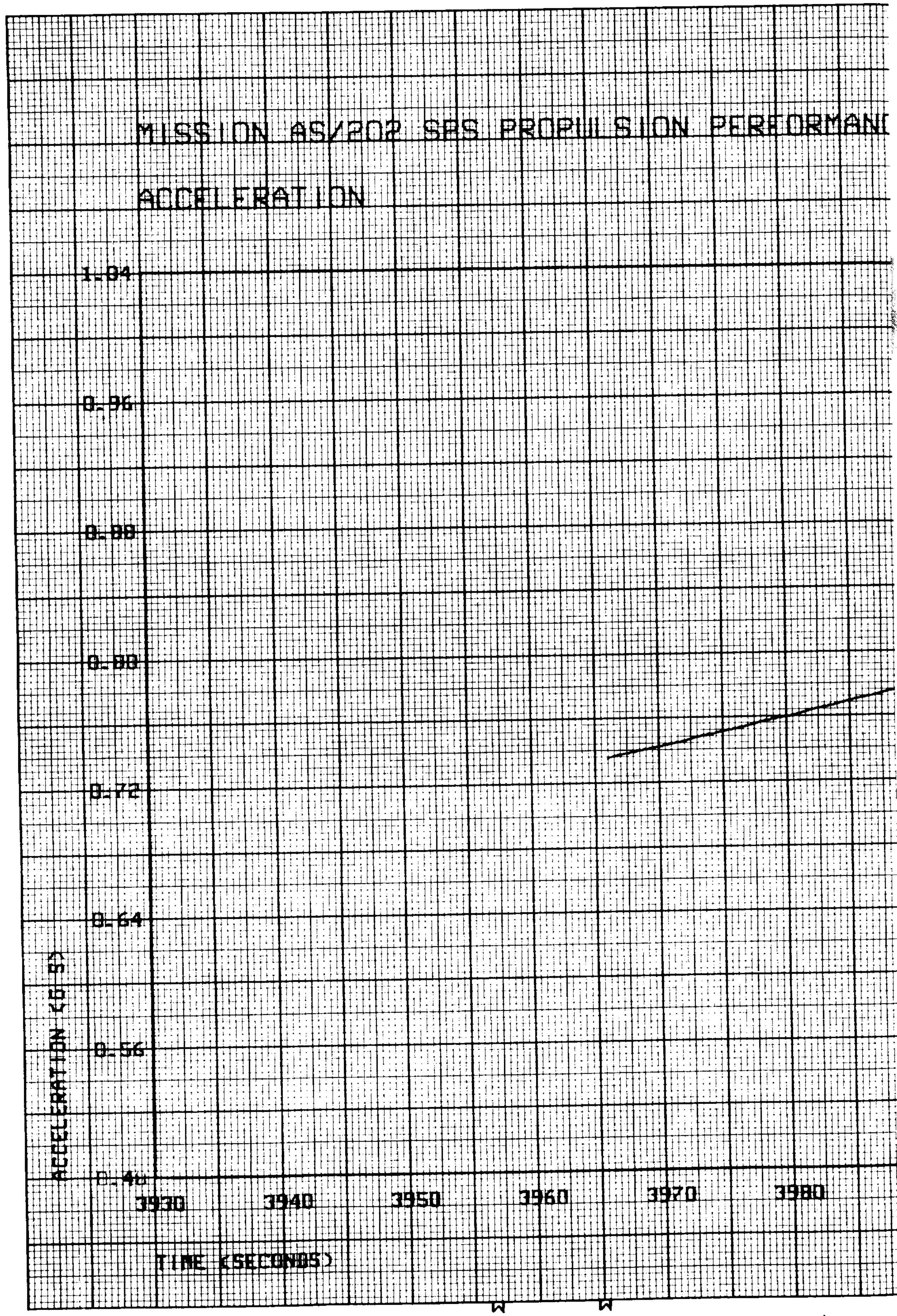
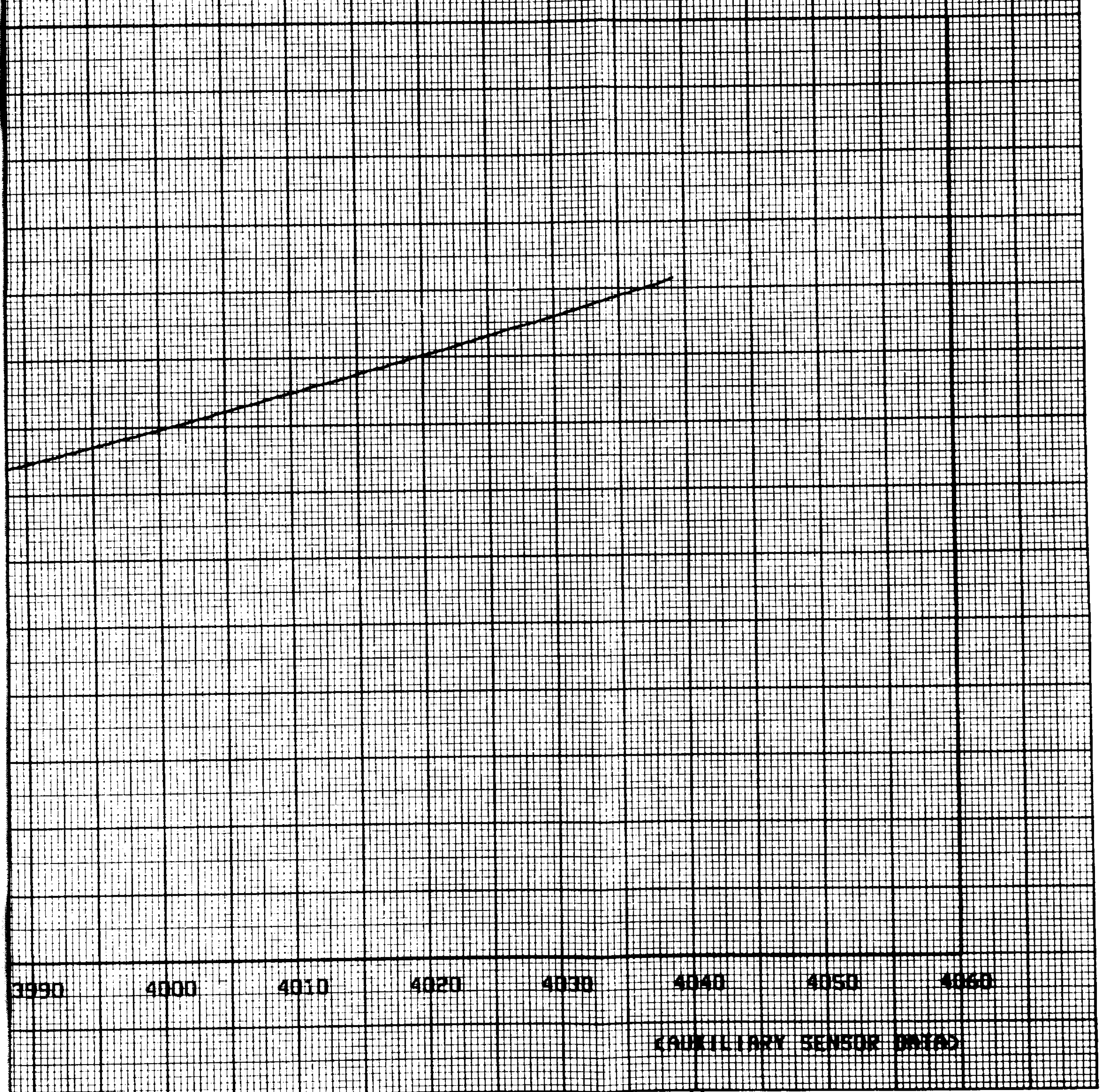


FOLDOUT FRAME

CE PARAMETERS SECOND BURN - 11/28

FIGURE 22



VII. REFERENCES

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2. TRW Systems Report 05952-H018-R8-00, "AS-202, Service Propulsion System, Quick Look Analysis Report (U)," prepared by the Apollo Propulsion Performance Evaluation Team, dated 1 September 1966. (C)

3. TRW Systems Report 3132.41-200, "Apollo, Mission 202, Spacecraft 011, ALDS Data," prepared by the Computation and Data Reduction Center.

4. TRW Systems Report 3132.41-202, "Apollo, Mission 202, Spacecraft 011, MCCK Data," prepared by the Computation and Data Reduction Center.

5. TRW Systems Report 3132.41-203, "Apollo, Mission 202, Spacecraft 011, Carnarvon Data," prepared by the Computation and Data Reduction Center.

6. TRW Systems Report 3132.41-204, "Apollo, Mission 202, Spacecraft 011, Carnarvon (Smoothed) Data," prepared by the Computation and Data Reduction Center.

7. TRW Systems Report 3132.41-205, "Apollo, Mission 202 Spacecraft 011, Antigua Data," prepared by the Computation and Data Reduction Center.

8. TRW Systems Report 3132.41-206, "Apollo, Mission 202, Spacecraft 011, Antigua (Smoothed) Data," prepared by the Computation and Data Reduction Center.

9. TRW Systems IOC 66:7224.2.6-6, "SPS Tailoff Velocity (AS-202)," prepared by Thomas H. Barrie, dated 31 October 1966.